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FLARED LANDING APPROACH FLYING QUALITIES

Volume I - Experiment Design and Analysis

Norman C. Weingarten, Charles J. Berthe, Jr.,
Edmund G. Rynaski, and Shahan K. Sarrafian

ARVIN/CALSPAN ADVANCED TECHNOLOGY CENTER
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FOREWORD

This report was prepared for the National Aeronautics and Space Administration, Langley Research Center, Hampton Virginia by Arvin/Calspan Advanced Technology Center, Buffalo, New York. It covers the preparation, conduct, and analysis of an in-flight simulation program investigating the flying qualities of aircraft in the final approach and flared landing task. The aircraft used was the USAF/AFWAL Total In-Flight Simulator (TIFS) which is operated by Calspan under Air Force Contract No. F33615-83-C-3603. The program was sponsored by NASA and administered by USAF/AFWAL. Additional funding was provided by the Boeing Commercial Airplane Company, Lockheed-Georgia Company, and the German Aerospace Research Establishment (DFVLR) to cover evaluation flights by test pilots from their respective organizations.

Mr. William D. Grantham was the project manager for NASA/Langley and Capt. Michael Masi was the TIFS Program Manager for AFWAL.

The work reported here was performed by the Flight Research Department of Calspan. Dr. Philip Reynolds was the TIFS program manager and Mr. Norman Weingarten was the project engineer. Messrs. Edmund G. Rynaski and Charles J. Berthe, Jr. were the principal investigators. Mr. Berthe was also the primary safety pilot and served as calibration pilot. Mr. Shahan Sarrafian of NASA/Dryden was the flight test engineer and also co-authored this report. Evaluation pilots were provided by NASA/Langley, NASA/Dryden, Calspan, as well as Boeing, Lockheed, and DFVLR.

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John Ball, Michael Parrag, Scott Buethe - Safety Pilots
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LIST OF SYMBOLS

ADI	attitude director indicator
b	wing span, ft
$\bar{c}$	mean aerodynamic chord, ft
CAP	control anticipation parameter = $\omega_{sp}^2/n_z/\alpha$
C_D	drag coefficient = $D/\bar{q}S$
C_{D_o}	drag coefficient at zero angle of attack
C_{D_i}	$\partial C_D/\partial i$, $i = \alpha, \delta_e$, deg $^{-1}$
c.g. or CG	center of gravity
C_L	lift coefficient = $L/\bar{q}S$
C_{L_o}	lift coefficient at zero angle of attack
C_{L_i}	$\partial C_L/\partial i$, $i = \alpha, \delta_e, \delta_{DLC}$, deg^{-1}
C_{L_j}	$\frac{2V}{\bar{c}}$ $\partial C_L/\partial j$, $j = \dot{\alpha}, q$, deg^{-1}
C_ℓ	rolling moment coefficient = $L/\bar{q}Sb$
C_{ℓ_i}	$\partial C/\partial i$, $i = \beta, \delta_a, \delta_r$, deg $^{-1}$
C_{ℓ_j}	$\frac{2V}{b}$ $\partial C\ /\partial j$, $j = p, r$, deg^{-1}
C_m	pitching moment coefficient = $M/\bar{q}S\bar{c}$
C_{m_o}	pitching moment coefficient at zero angle of attack
C_{m_i}	$\partial C_m/\partial i$, $i = \alpha, \delta_e$, deg $^{-1}$
C_{m_j}	$\frac{2V}{\bar{c}}$ $\partial C_m/\partial j$, $j = \dot{\alpha}, q$, deg^{-1}
C_n	yawing moment coefficient = N/qSb
C_{n_i}	$\partial C_n/\partial i$, $i = \beta, \delta_a, \delta_r$, deg $^{-1}$
C_{n_j}	$\frac{2V}{b}$ $\partial C_n/\partial j$, $j = p, r$, deg^{-1}
C_y	side force coefficient = $Y/\bar{q}S$
C_{y_i}	$\partial C_y/\partial i$, $i = \beta, \delta_a, \delta_r$, deg $^{-1}$
C_{y_j}	$\frac{2V}{b}$ $\partial C_y/\partial j$, $j = p, r$, deg^{-1}

LIST OF SYMBOLS (Cont'd)

CR	center of rotation
D	drag, lb
DLC	direct lift control
D.R.	digital reversion mode
dB	decibel units for Bode amplitude = 20 $\log_{10}$ (amplitude)
F_{AW}	aileron wheel force, lb
F_{ES}	elevator wheel force, positive aft, lb
F_{RP}	rudder pedal force, lb, positive, right pedal forward
g	gravitational constant = 32.17 ft/sec^2
h	altitude of airplane c.g., ft
h_p	altitude of airplane at pilot station, ft
h_{WH}	altitude of airplane at model wheels, ft
HQR	Handling Qualities Rating
HSI	Horizontal Situation Indicator
I_{xx}, I_{yy}, I_{zz}	moments of inertia about X, Y, Z body axes, slug-ft^2
I_{xz}	product of inertia about X, Z body axes, slug-ft^2
IFR	Instrument Flight Rules
K_{P_h}	steady state pilot gain altitude loop closure, rad/ft
K_{P_θ}	steady state pilot gain in attitude loop closure, lb/rad
K_q	loop gain in pitch augmentation system, deg/deg/sec
K_C	command gain
K_I	integrator gain
K_L	loop gain
L	lift, lb
L, M, N	moments about X, Y, Z body axes, ft-lb

m	mass of airplane, slugs
ms	milliseconds
N.D.	normal digital mode
n_y, n_z	lateral, normal acceleration, g's (n_y + right, n_z + down)
n_j^i	transfer function numerators
N_q, N_α, N_V	transfer function numerators of q, α, v per δ_e
$N_{z_{CG}}$	normal acceleration at CG
N_{z_p} or n_{z_p}	normal acceleration at pilot station
p, q or Q, r	roll, pitch, yaw rates, deg/sec
PIOR	pilot-induced oscillation rating
PR	pilot rating
PSD	power spectral density
$\bar{q}$	dynamic pressure = $\frac{1}{2}\rho V^2$, lb/ft^2
q_c	pitch rate command
s	Laplace operator, sec^{-1}
S	reference wing area, ft^2
T	total thrust, lb
T_D	equivalent time delay from equivalent systems analysis, sec
T_z	lead + lag pre-filter zero
T_F	lead + lag pre-filter pole
T_{WO}	washout pre-filter pole
t_1	effective time delay from maximum slope intercept method, sec
$1/\tau_{\theta_1}$	lower frequency zero in q/δ_e transfer function
$1/\tau_{\theta_2} \approx L_\alpha$	higher frequency zero in q/δ_e transfer function, $\frac{g}{V}\ \frac{C_{L_\alpha}}{C_L}$
TIFS	Total In-Flight Simulator

Δt	rise from time history criteria analysis, sec
V	true airspeed, ft/sec
V_I	inertial speed, ft/sec
V_x, V_y, V_z	velocity components along X, Y, Z body axes, ft/sec
W	airplane weight, lb
X, Y, Z	body axes, X-Z plane is in plane of symmetry with X directed forward parallel to the fuselage reference line, Z directed downward, and Y directed out the right wing
X_{MP}	distance along X-body axis between c.g. and pilot station, ft
X_{PCR}	distance along X-body axis between instantaneous center of pitch rotation and pilot station, ft
Y_{P_h}	pilot describing function in altitude loop closure
Y_{P_θ}	pilot describing function in attitude loop closure
α	angle of attack, deg
α_g	turbulence component of angle of attack, deg
α_T	total angle of attack with respect to true airspeed, deg
α_I	inertial angle of attack (with respect to inertial velocity), deg
β	sideslip, deg
β_g	turbulence component of sideslip, deg
β_T	total sideslip with respect to true airspeed, deg
β_I	inertial sideslip (with respect to inertial velocity), deg
γ	flight path angle, deg
Δn_z	incremental normal acceleration, g's
δ_a	aileron surface deflection, included angle positive left T.E. down, deg
δ_{AW}	aileron wheel deflection, positive clockwise, deg

δ_{AWC}	aileron command positive left T.E. down, deg
δ_e	elevator surface deflection, positive T.E. down, deg
δ_{EC}	elevator column deflection, positive aft, inch
δ_r	rudder surface deflection, positive T.E. left, deg
δ_{RP}	rudder pedal deflection, positive right pedal forward, inch
δ_{RPC}	rudder, command positive T.E. left, deg
δ_{th}	throttle lever position, deg
ζ	damping ratio
ζ_d	damping ratio of Dutch roll mode
ζ_{ph}	damping ratio of phugoid mode
ζ_{sp}	damping ratio of short period mode
ζ_{rs}	damping ratio of coupled roll-spiral mode
θ	pitch attitude, deg or rad
λ	aperiodic real root magnitude, sec^{-1}
ρ	air density, $slug/ft^3$
σ_i	mean square gust intensity, $i = \alpha, \beta$, deg
τ_L	time constant of pilot's lead element, sec
τ_R	time constant of roll mode, sec
τ_S	time constant of spiral mode, sec
ϕ	bank angle, deg
ω_{BW}	bandwidth frequency, rad/sec
ω_d	undamped natural frequency of Dutch roll mode, rad/sec
ω_{ph}	undamped natural frequency of phugoid mode, rad/sec
$\omega_{n_{sp}}$	undamped natural frequency of short period mode, rad/sec
ω_{rs}	undamped natural frequency of coupled roll-spiral mode, rad/sec

LIST OF SYMBOLS (Cont'd)

SUBSCRIPTS

c.g.	center of gravity
DLC	direct lift control
e	equivalent parameter from equivalent system analysis
g	turbulence component
G.E.	ground effect
I	inertial quantity
m	model quantity
MP or PM	model quantity at pilot station
MTCG	model quantity transformed to TIFS c.g.
P	quantity at pilot station
TIFS or unsubscripted	TIFS quantity at its c.g.
sp	short period mode
WH	model wheel height
$m_{c.g.}$	model center of gravity
stab	stability axis

Section 1
INTRODUCTION

Much research using in-flight simulators has been conducted in recent years in the area of flying qualities for the flared landing approach phase of flight (References 1 through 5). Most of these studies dealt with obtaining data to verify or refine old criteria and to develop new criteria. Because of the introduction of highly augmented aircraft with the resulting non-conventional airplane response, effective time delay, and higher-order dynamic behavior, some flying quality criteria that concentrated on one airplane state, namely pitch rate, have been shown not to work well. The primary emphasis of this in-flight flying qualities experiment utilizing the USAF/TIFS (Total In-Flight Simulator), was to generate a consistent set of data to find out what the pilot requires in order to be able to flare and land an airplane. Two separate areas of analysis were performed on the data. One was to investigate what kind of commanded response (e.g., angle of attack or pitch rate) and its characteristics that the pilot preferred. The other area was to refine a time history criterion that took into account all the necessary variables and their characteristics that would accurately predict flying qualities. The result of the first part of the program would be to provide guidelines to the flight control system designer in developing systems using MIL-F-8785(C) as a guide that would yield the responses that pilots prefer in flared landings. The second part of the program provides the flying qualities engineer with an accurate predictive tool which would tell him how good the resulting system would be.

Flight Control Design Configurations

The flying qualities specification published in 1969, MIL-F-8785(B) (Reference 6) represented the culmination of nearly 20 years of experimental flight research involving many variable stability aircraft such as the USAF/Calspan NT-33A, the Princeton Navion, and the NASA/Boeing 367-80. This specification for the first time quantitatively defined satisfactory and acceptable regions of specific modal parameters of an airplane, such as short period, phugoid and Dutch roll frequencies and damping ratios. A more recent revision, MIL-F-8785(C), (Reference 7) for the most part preserves the modal requirements specified in MIL-F-8785(B) without significant revision both in terms of the modal requirements and definitions of the modes. MIL-F-8785(C), for instance, states that "short period response of angle of attack shall meet the requirements ...". The proposed MIL Standard and Handbook (Reference 8), however, is oriented almost totally in the direction of defining an "equivalent" short period mode from the pitch rate rather than the angle of attack response of the vehicle. If the vehicle is unaugmented or if the response order has not been increased by compensation or other dynamic elements, then it does not matter whether the short period is defined from the angle of attack or from the pitch rate behavior of the vehicle. However, if dynamic elements have been introduced in the control system, such as a pre-filter or integral plus proportional compensation in the loop with pitch rate feedback, then it is unlikely that the short period frequency, equivalent or otherwise, may be accurately obtained from the pitch rate response of the vehicle.

The flying qualities specification, MIL-F-8785(C) and the preceding (B) version, unfortunately have seldom served as direct criteria for the design of a flight control system for the longitudinal-vertical degrees of freedom of motion of an aircraft. There appears to be two primary reasons for this:

1. The effect of compensation networks and other dynamic elements that produce higher order response effects in the flying qualities frequency range of interest were not explicitly addressed in the specification. The implicit message of the specification is that higher order response behavior almost always degrades the flying qualities of the vehicle.

2. The flight control system designer is oriented more towards the command input-controlled output response philosophy. Generally speaking, the flight control system designer has often considered the feedback quantities to be the "controlled variable(s)" and seeks design criteria based upon the response of the "controlled variables" to a command input. The modal approach of MIL-F-8785(C) addresses dynamic requirements in terms of commanding the entire airplane rather than a specific

state variable(s) of the airplane. Although the command-response approach to control system design is not incompatible with the requirements, the tendency has been to generate new criteria based upon the feedback rather than the modal parameter requirements. Several significant examples of this approach to control system criteria exist. The most prominent among them are C*(t) (Reference 9), the Shuttle Orbiter pitch rate envelope criteria (Reference 10), and the angle of attack time history response envelope (Reference 11).

There has been a profusion of alternate flying qualities criteria since the publication of MIL-F-8785(B). Criteria are open-loop, closed-loop, man-in-the-loop and both frequency domain and time domain oriented. Some attempt directly to transform from the modal specification format of MIL-F-8785(C) while others are empirically derived from the analysis of sets of flight data. Although most are successful to some degree, they are primarily compliance or evaluation methods rather than direct design criteria that can aid the designer in trying to decide what to feedback, how much and whether higher order networks such as an integral plus proportional compensation should be added to the system. All of these criteria can be useful, but most do not appear to have the objective of direct extensions to the solid foundation of MIL-F-8785 (C) or interpretation of MIL-F-8785(C) for the flight control system designer. It would seem to be a desirable ultimate objective to help the control system designer make direct use of the results of the twenty years of experimental flight test that are embedded in the flying qualities specification.

It is an objective of the experimental flight program described in this report to interpret the flying qualities requirements of MIL-F-8785(C) in terms of command/response configurations. In this sense, then, the effort described in this report is directed toward the first steps required to transform MIL-F-8785(C) from a pure flying qualities specification into a joint flying qualities/flight control design criteria. The two commanded longitudinal state variables chosen for investigation are pitch rate and angle of attack, the two "natural" states of constant speed aerodynamic flight. By evaluating pitch rate command and angle of attack command configurations a few basic guidelines are established to help the flight control system designer decide whether the system should behave as a conventional aircraft (angle of attack command system) or as a rate command, attitude hold system. Rate command and angle of attack command configurations were specified totally within the context of MIL-F-8785(C) in order to determine pilot preference for one or the other as a function of $1/\tau_{\theta_2}$ and phugoid mode characteristics. The results suggest that pilot preference for angle of attack or pitch rate command is a function of $1/\tau_{\theta_2}$.

Time Domain Criteria (Pitch Axis, Flared Landing Task)

The primary objective of this phase of the experiment was to refine the time domain predictive criteria of Reference 5. (This criteria is applicable to the pitch axis in the flared landing task.) A major limitation of the original criteria was an inability to contend with time delays. An additional limitation was concern as to the ability to handle excursions of short period frequency that fell outside the borders of Level 1 boundaries of MIL-F-8785 N_z/α vs. $\omega_{n_{sp}}$ requirements.

Experience with the Calspan Learjet had indicated that time delay effects were strongly dependent on control sensitivity (sensitivity defined, in this instance, as maximum slope of the pitch rate response per pound of pitch controller input). Consequently, it was determined that a valid time delay experiment must include sensitivity effects and these sensitivity effects might be applicable to high frequency short period cases from the MIL-F-8785 requirements.

A time delay test matrix was chosen which utilized two baseline flight configurations that were Level 1 in flying qualities performance (one a "conventional" angle of attack command, and the other a pitch rate command). The matrix was completed by applying three levels of pitch sensitivity and three levels of time delay to the baseline configurations.

The in-flight data provided by this matrix resulted in the desired time delay and sensitivity matrices for criteria refinement. The resultant refined criteria was applied to applicable configurations of this experiment as well as to the flight data of the pitch rate program (Reference 5), the

LAHOS program (Reference 1), the Large Aircraft program (Reference 4), the Ames NT-33 Study (unpublished), the SST program (Reference 12), and the X-29 program (unpublished), for a total of 129 configurations. These 129 configurations consisted of variations in short period frequency and damping, angle of attack command, pitch rate command, $1/\tau_{\theta_2}$ variations, time delay variations, prefilter variations, sensitivity variations, wheel and stick controllers, and aircraft sized from the X-29 to 1,000,000 lb. gross weight.

The results were that the time domain predictive criteria predicted 60% of the 129 configurations from seven different programs to within one Cooper-Harper rating, 88% to within two HQR ratings, and 81% by flying qualities Level. The predictions for this subject program, where seven pilots and many repeated evaluations provided a more valid statistical base, were: 77% within 1 HQR, 96% within 2 HQR's, and 100% by flying qualities Level.

A number of frequency domain predictive techniques were applied to the data of this program and to selected configurations from other programs. The most promising of these was a technique using Neal-Smith pilot lead compensation angle as the prime parameter. For this program the results were: 65% predicted within ±1 HQR, 81% within 2 HQR and 73% by flying qualities level, however, 19% missed by more than 2 HQR's (compared to 4% for the time domain criteria) and 4 Level 3 configurations were missed (compared to none missed by the time domain criteria). This technique did not successfully account for time delay and sensitivity effects and needs to be applied to a wider data base.

The balance of the report is organized in two volumes as follows:

<u>Volume I</u>

Section 2 — Experiment Design - includes the detailed configuration descriptions and evaluation procedures.

Section 3 — Experiment Mechanization - describing the TIFS aircraft and equipment.

Section 4 — Data - includes the flight chronology and raw data obtained in the experiment (with some references to the Appendices).

Section 5	Interpretation of the Results Using MIL-F-8785(C) as a Flight Control Design Criteria.
Section 6	Predictive Criteria Results and Analysis - includes time domain and frequency domain analyses.
	Sections 5 and 6 each contain their own conclusions and recommendation subsections.
Section 7	References

Volume II

Appendix A	Pitch Step Responses
B	Throttle Step Responses
C	Model Following Verification Steps
D	Pilot Comments
E	Approach Time Histories
F	Neal-Smith Parameter Plane Plots

Section 2
EXPERIMENT DESIGN

The motivation for the flight experiments performed using the USAF/AFWAL Total In-Flight Simulator (TIFS) and discussed in this report are many. The flying qualities specification MIL-F-8785(C) presents flying qualities requirements in terms of satisfactory and acceptable regions of specific modal parameters of an airplane such as short period and phugoid frequency and damping ratio. Modal residues (or zeros of transfer functions) are not specifically addressed, therefore, the responses between aircraft having the same modal characteristics can vary over a wide range.

This experiment explores, in a preliminary way, a range over which the response of the aircraft may vary yet satisfy flying qualities requirements as defined by MIL-F-8785(C). The range of dynamic behavior is related to flight control system design considerations in the sense that purely angle of attack command and purely pitch rate command configuration are compared in as directly a form as possible. For instance, all the configurations were designed to have a short period natural frequency of 2 rad/sec. For the angle of attack configuration, the short period damping ratio was held constant at $\zeta_{sp} = 0.7$. The pitch rate command configuration required, because of the "command" requirements, variations in ζ_{sp} as a function of $1/\tau_{\theta_2}$. A more complete discussion of the rationalization for the selection of these configurations is given in Reference 13.

The flight control system designer is often oriented toward the command/response philosophy of control system design, so the experiments are designed to provide, in a preliminary sense, an interpretation or guideline for the control system designer in the sense that he will be able to decide whether to design a control system that will result in an angle of attack command (or conventional aircraft) responding system or a pitch rate command, attitude-hold system. In this sense, the attempt is to demonstrate to the flight control system designer that not only are the specifications in MIL-F-8785(C) an appropriate way to judge compliance with flying qualities requirements, but it may also be directly used as a flight control system design criteria. In this way, the results of the twenty years of experimental flight tests used to define the flying qualities specification can also be directly used by the flight control system designer.

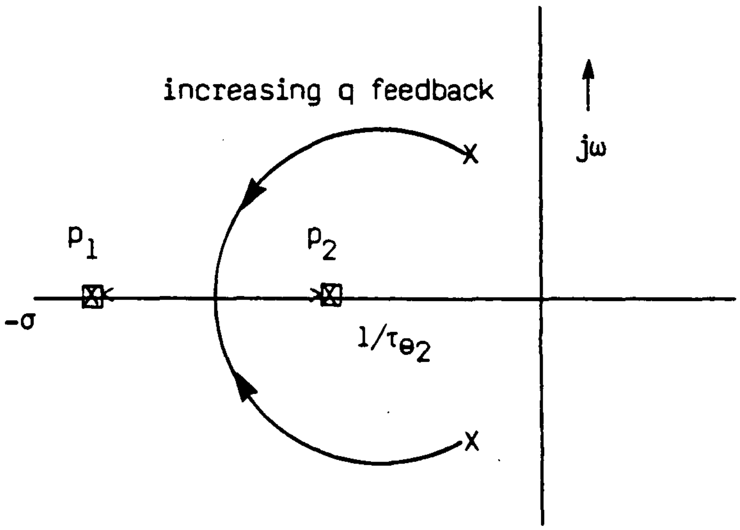
2.1 CONFIGURATION DEFINITIONS

The experiment described below considers angle of attack and a pitch rate command systems in terms of the locations of the poles of the system with respect to the zeros in various transfer functions. In this respect, the intent is to try to determine whether the modal parameter approach as specified in MIL-F-8785(B) has been properly interpreted in the MIL-F-8785(C) standard in terms of aircraft axes. The short period and the phugoid will be considered separately because it is possible to design a pitch rate command system for the short term, but an angle of attack command system in the long term, or vice versa. The idea is to try to determine pilot preference both in the short term and long term.

The angle of attack or the pitch rate command systems can be defined solely in terms of the locations of the vehicle poles with respect to the zeros of the transfer functions. A pitch rate command, attitude hold system will produce a pole-zero cancellation such that three poles are placed at the zeros of the transfer function located at the origin of the s plane, at $-1/\tau_{\theta_1}$, and at $-1/\tau_{\theta_2}$. Therefore, the response in pitch rate is dominated by the one remaining pole. In the angle of attack command system two poles are located at the low frequency zeros ω_α, ζ_α of the angle of attack transfer function. The response is dominated by the remaining two poles, which define the short period mode. These systems are briefly described below without regard to how they may be mechanized. The mechanization problem is not a difficult one and is discussed in Section 2.2.1 and Reference 13.

2.1.1 Configuration Selection Rationale

For this experiment, command configurations such as pitch rate, angle of attack, or rate of change of flight path were designed with direct reference to MIL-F-8785(C). The configurations were defined as if the particular quantity commanded or fed back was constrained by the requirements of the modal parameters of the MIL-F-8785(C) format. Consider the sketch below, which shows the root locus plot for a two degree of freedom vehicle representation using pitch rate feedback.



PITCH RATE FEEDBACK - 2 DOF VEHICLE REPRESENTATION

As the feedback gain is increased, the damping ratio of the short period mode becomes high and eventually the short period mode separates into two real roots; one that will terminate at the zero located at $s = -1/\tau_{\theta_2}$ and the other at infinity <u>if not constrained</u>. The short period mode is defined by $(s + p_1)(s + p_2)$ and indicates a damping ratio greater than critical with short period frequency of $\omega_{sp} = \sqrt{p_1 p_2}$ and damping ratio of $\zeta = \dfrac{(p_1 + p_2)}{\sqrt{p_1 p_2}}$.

Therefore, in the limit one of the short period poles will be located at $s = -1/\tau_{\theta_2}$ and the pitch rate response will be a pure first order response characterized by the transfer function $q/F_s(s) = K/(s + p_1)$, even though the short period mode itself is characterized by $(s + p_1)(s + 1/\tau_{\theta_2})$.

2.1.2 Pole-Zero Patterns

The sketches below show the pole-zero pattern representation of the different "controlled variable" configurations.

Angle of Attack Command System (Configurations 1, 5)

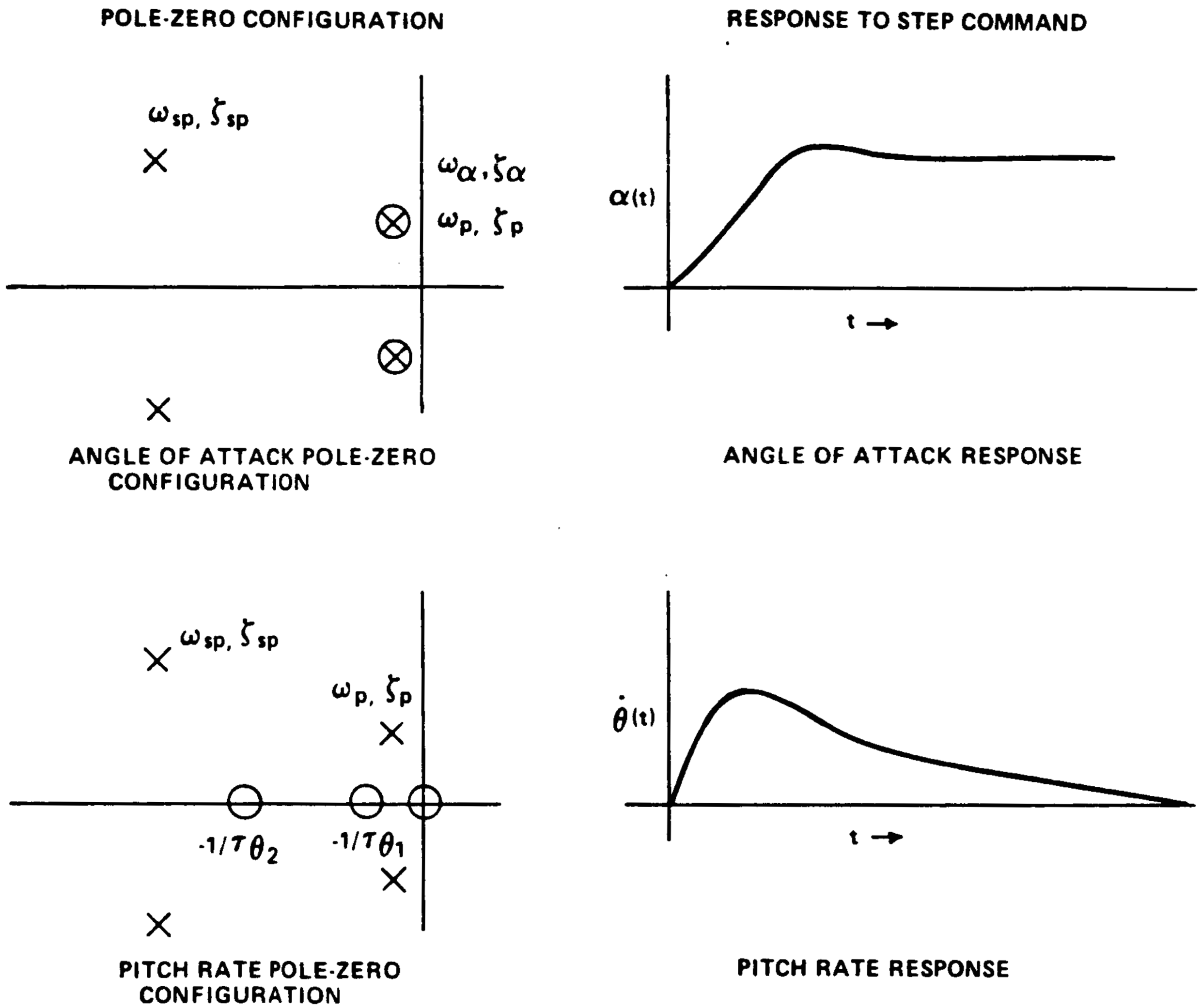


Figure 1. Angle of Attack Command System

As shown in Figure 1, the response of the angle of attack command system is dominated by the short period poles ω_{sp}, ζ_{sp}. The phugoid poles are located at the low frequency zeros ω_α, ζ_α of the $\alpha/\delta_F(s)$ transfer function. The result can be a quick, smooth and well behaved angle of attack response as defined by the short period mode. Theoretically there is no residue in the angle of attack response in the phugoid mode; i.e., $\dot{\alpha}(t) = 0$ after the short period response.

The pitch rate response of the angle of attack command system is typical of a conventional aircraft. The transfer function zero at $-1/\tau_{\theta_2}$ produces an overshoot in the pitch rate response to a step command input, and a significant phugoid mode residue with zero ultimate steady state value is evident.

<u>Pitch Rate Command System</u> (Configuration 2, 6)

As the angle of attack command system showed pole-zero cancellation in the angle of attack transfer function, the pitch rate command system indicates pole-zero cancellation in the pitch rate transfer function. The pole-zero pattern showing these cancellations are indicated in Figure 2 below.

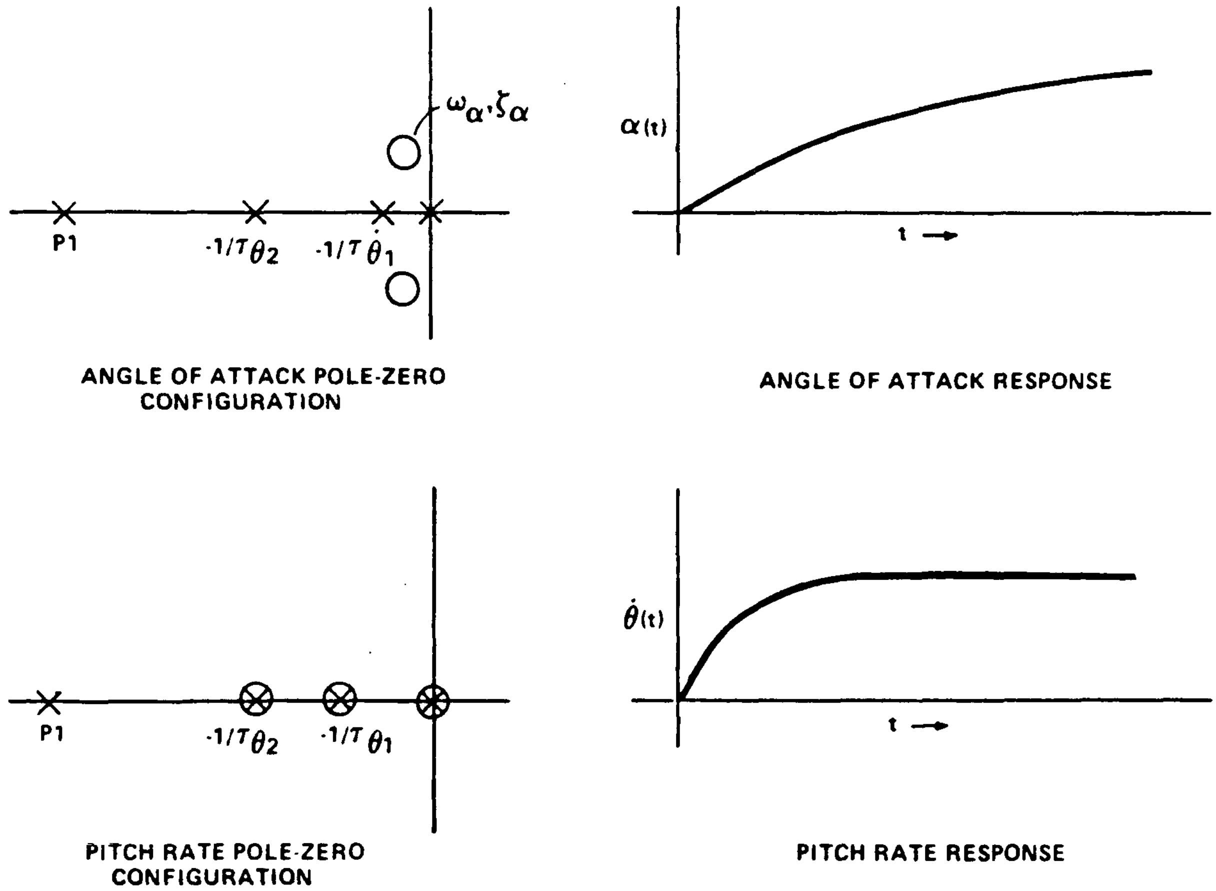


Figure 2. Pitch Rate Command System

The response of pitch rate in the pitch rate command system is dominated by the single order pole p_1 shown in the above figure. The response shows no residue in the phugoid mode, and the zero at the origin is cancelled by a pole, which indicates that the system will be an attitude-hold system. The angle of attack response is generally sluggish, dominated by the poles at $-1/\tau_{\theta_2}$ and $-1/\tau_{\theta_1}$ that are not cancelled by numerator zeros of the angle of attack transfer function. The pole at the origin also contributes to the response and leads to a steady state ramp response in angle of attack. After the step input is returned to zero, the pitch rate returns to zero but the change in angle of attack does not. The pitch rate command, attitude hold system is also an angle of attack "hold" system, although the response in angle of attack is normally so sluggish that steady state angle of attack would likely be rarely seen in actual flight. Speed change will also exhibit neutral stability.

The two types of system described in Figure 1 and Figure 2, mainly a pitch rate command and angle of attack command system involve both short period and phugoid dynamic behavior of both of the response variables. Each part contributes significantly to the dynamic behavior of the system. The low frequency behavior of the angle of attack command system is such that after the angle of attack reaches steady state, then changes in flight path are equal to changes in pitch angle; i.e., $\Delta\gamma \doteq \Delta\theta$ since $\dot{\alpha}(t) \doteq 0$. The pilot can judge changes in flight path by observing changes in pitch angle. Because pitch rate eventually goes to zero following a step command, the pitch attitude and the flight path reach new steady state values. In the pitch rate command, attitude-hold system, the angle of attack responds sluggishly and never reaches a steady state value to a step command input. The change in flight path angle is not equal to changes in pitch angle; i.e., $\Delta\theta \neq \Delta\gamma$ and the pilot has more difficulty in judging changes in flight path by observation of changes in pitch angle. The result of the sluggish angle of attack response is frequently an overcontrol tendency by the pilot during flare and landing. A correction of the overcontrol leads to pilot complaints of "non-monotonic" stick forces.

The differences in the short period response are more obvious. In the angle of attack command system, the numerator zero in the pitch rate transfer function may be considered a lead term in the pitch rate response. In the pitch rate command system, the singularity that previously was a pitch rate lead becomes a pole or lag in the angle of attack response.

Hybrid Systems

Simple variations in the types of pure controlled variable systems designed for this experiment should allow both the flying qualities engineer and the flight control system designer to determine whether or not the "controlled variable" philosophy of control system design applies to both the short term and the long term or phugoid mode. For instance, by the independent placement of the short period and phugoid poles, it is a relatively simple matter to obtain a short term angle of attack command, long term pitch rate command system. This can be done as shown in Figure 3 below, in which the short period poles are placed as if the system were angle of attack command, while the phugoid poles are placed as if the system were pitch rate command. The converse, as shown in Figure 4 below, can also be accurately evaluated using the TIFS variable stability airplane.

In the past it has been often stated that the pilot is little affected by the long term or phugoid motion of the vehicle. It has been assumed that the pilot either ignores these long term effects or corrects for them more-or-less subconsiously. If this hypothesis is true, then it should make no difference if the phugoid poles were located at either the zeros of the numerator of the angle of attack transfer function (ω_α, ζ_α) or at the origin and at $-1/\tau_{\theta_1}$, two of the numerator zeros of the pitch rate transfer function. It is expected that the hybrid variations depicted by Figures 3 and 4 whould help significantly to settle the question of the importance of phugoid dynamics with respect to flying qualities.

Short Term Angle of Attack - Long Term Pitch Rate (Configuration 3, 7)

The short-term angle of attack, long-term pitch rate command system pole-zero configuration is shown in Figure 3.

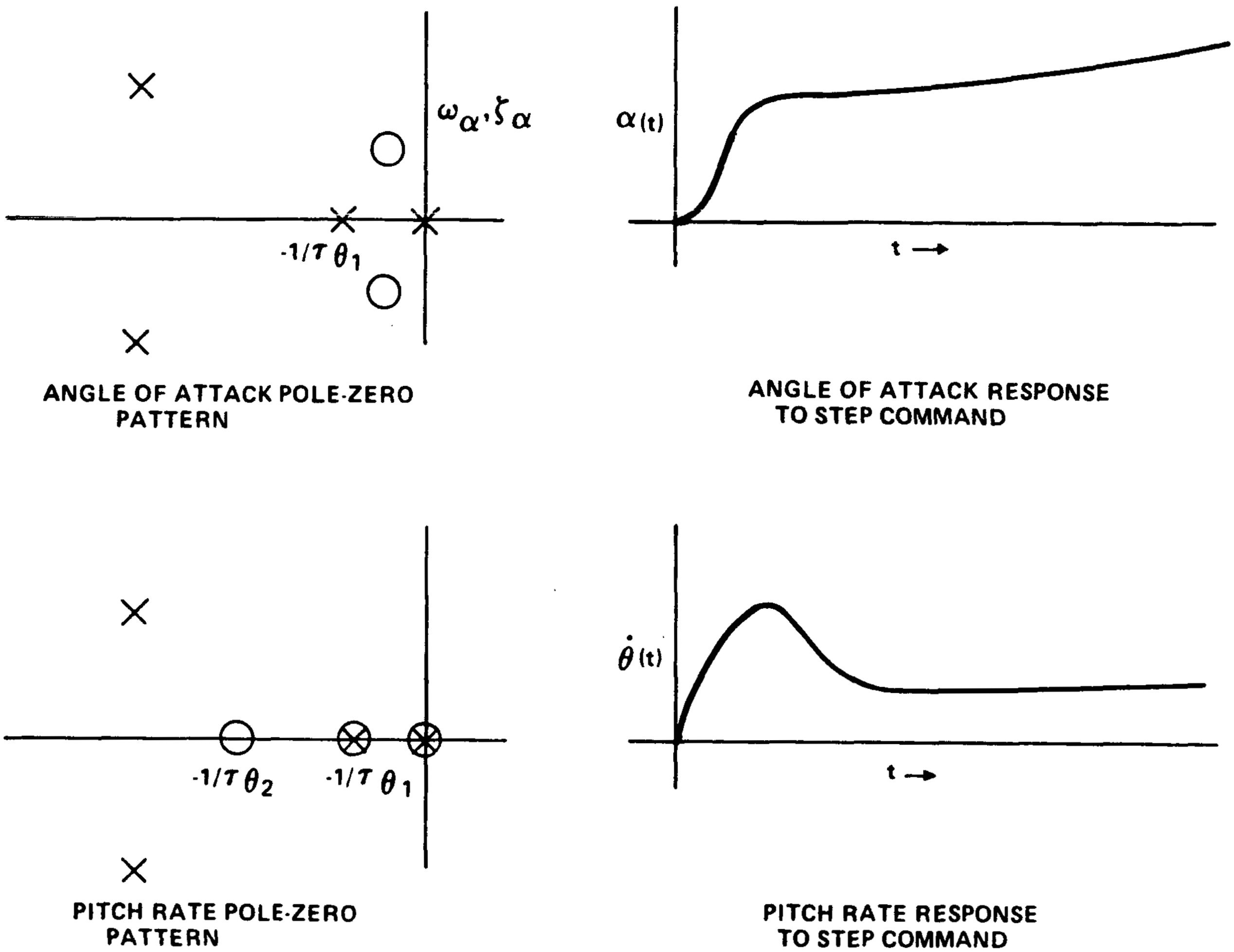


Figure 3. Short Term Angle of Attack, Long Term Pitch Rate Command System

The behavior of this system is characterized by the smooth and rapid short period angle of attack response and the modal residues of the poles located at $1/\tau_{\theta_1}$ and at the origin. The pitch rate response is characterized by an initial pitch rate overshoot followed by a steady state pitching rate; no phugoid mode residue is evident.

Short Term Pitch Rate - Long Term Angle of Attack Command System
(Configuration 4, 8)

The short term pitch rate, long term angle of attack system is shown in Figure 4 below:

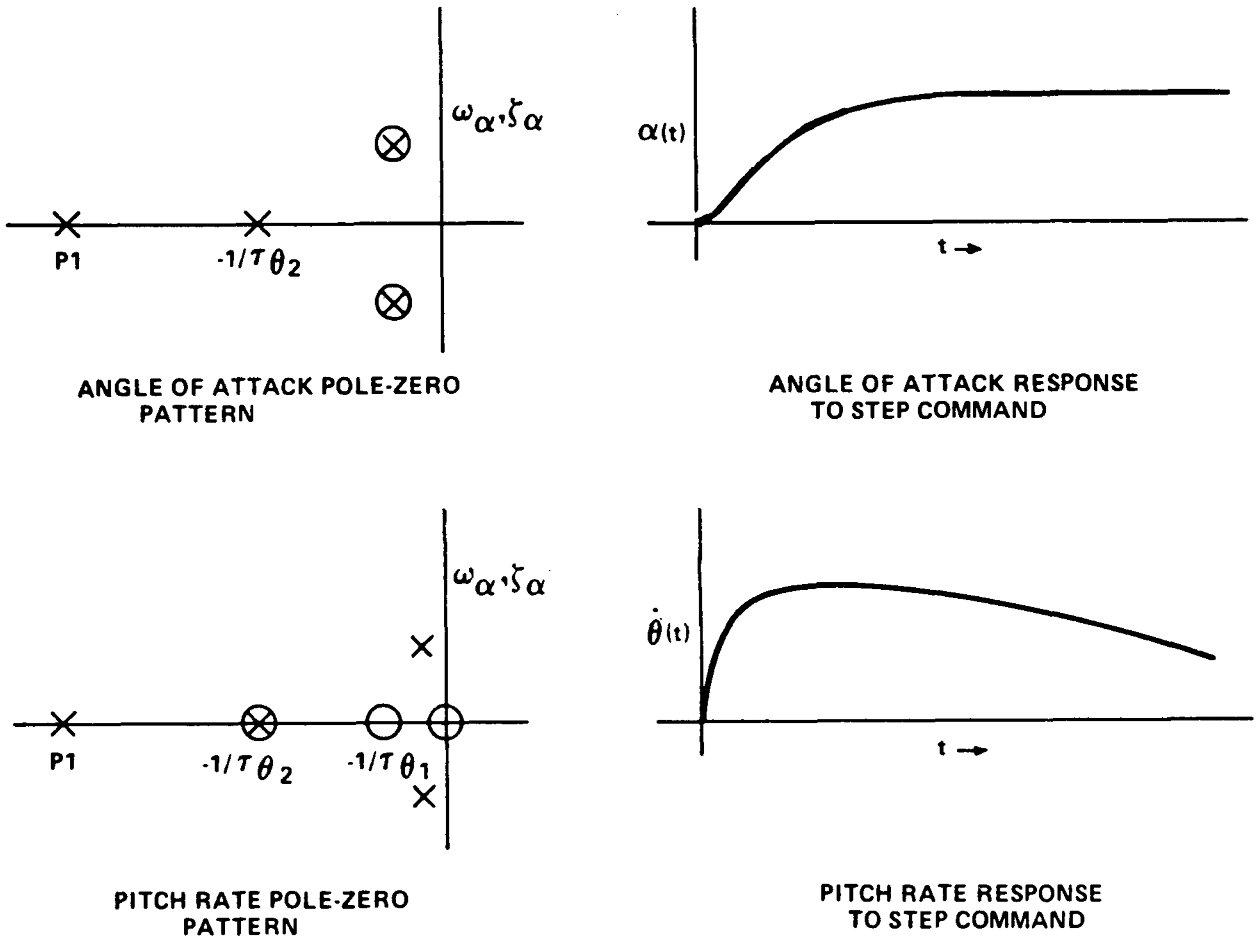


Figure 4. Short Term Pitch Rate, Long Term Angle of Attack Command System

The behavior of this system is characterized by an angle of attack response dominated by the pole at $-1/\tau_{\theta 2}$ and can be sluggish. The angle of attack response remains steady in the long term. The pitch rate response is initially rapid and dominated by the single pole at $-p_1$, but then exhibits the effects of a significant residue at the phugoid mode frequency.

2.1.3 Configuration Parameter Selection

All configurations with one exception, Configuration 12, were deliberately designed with the pilot located at the center of rotation (i.e., percussion) in order to eliminate the initial acceleration experienced by the pilot during maneuvering when located somewhere other than the center of rotation. For all configurations the short period natural frequency was held constant at ω_n = 2 rad/sec, while $1/\tau_{\theta_1}$ was held constant at $1/\tau_{\theta_1} = 0.1$ rad/sec and the high frequency zero of the $\alpha/F_S(s)$ transfer function was located for all configurations at $1/\tau_\alpha = -3.7$. This value was relatively low but yielded a vehicle configuration having the pilot and center of rotation co-located.

The first four configurations 1-4 were chosen to have a value of $1/\tau_{\theta_2} = 0.5$ and the low frequency numerator zeros of the angle of attack transfer function were defined to have a frequency ω_α of 0.3 rad/sec with $\zeta_\alpha = 0.10$. For the pitch rate command configuration, the result was a very rapid response in pitch rate dominated by a pole at s = -8.0 while the angle of attack response was sluggish, dominated by the pole at s = -0.5 (or $-1/\tau_{\theta_2}$). The zeros of the angle of attack transfer function meant that large phugoid mode residues would appear in the attitude and flight path response for the angle of attack command configurations. The purpose of these designs were to test the following hypothesis:

1. Although the pitch rate response could be very rapid, the vehicle may not be Level 1 if the angle of attack response in the short term is sluggish. The purpose is to demonstrate that the MIL-F-8785(C) short period requirements pertain to the angle of attack response of the vehicle.

2. The phugoid mode may be relatively high frequency and low damping ($\omega_{ph} = 0.3$, $\zeta_{ph} = 0.1$) and still be acceptable if the vehicle is configured as an angle of attack command system. The large phugoid mode residues appearing in the attitude and flight path responses are acceptable to the pilot because the angle of attack response is well behaved in the long term, $\alpha(t)$ exhibits no phugoid mode residue and, therefore, $\Delta\theta = \Delta\gamma$ in the long term.

Configurations 5-8 were designed to have a value of $1/\tau_{\theta_2} = 0.9$ and low frequency angle of attack transfer function zeros defined by $\omega_\alpha = 0.1$, $\zeta = 0.1$. The short term pitch rate response of the pitch rate command configurations would be slower than those of Configuration 2 or 4, but the angle of attack response would be more rapid in the short term. The lower phugoid mode frequency located at a frequency of 0.1 rad/sec is identical to the value of $1/\tau_{\theta_1}$. Therefore, the phugoid mode residue in the pitch attitude and flight path angle responses would be small for all the Configuration 5-8. The purpose of these selections were to test the following hypothesis:

1. A pitch rate command configuration would be rated Level 1 if the angle of attack response satisfied the ω_n vs n/α requirement of MIL-F-8785(C). If true, the flying qualities of a pitch rate command system would be a direct function of the value of $1/\tau_{\theta_2}$.

2. The flying qualities of a pitch rate command system would improve if changes in attitude more closely corresponded with changes in flight path angle in the long term. This is manifested by a smaller long term residue in the angle of attack response.

Configurations 9 and 10 were designed to have no phugoid residue in either the pitch rate or angle of attack response in the long term. The low frequency zeros of the $\alpha/F_S(s)$ were given values equal to those of the $q/F_S(s)$ transfer functions, i.e., $s = 0, -1/\tau_{\theta_1}$. Therefore, the system would be both angle of attack and pitch attitude "hold" in response to pilot commands. This kind of configuration is feasible with an additional control surface such as a canard surface with relatively low power and low bandwidth characteristics. The neutrally stable phugoid response would appear only in the speed change degree of freedom of motion. The purpose of these configurations is to test the following:

1. The pilot would find quite acceptable, and even prefer a vehicle that exhibited no phugoid mode behavior in both angle of attack and pitch rate.

2. By eliminating the phugoid mode to stick command inputs, the importance of the phugoid mode to flying qualities can be demonstrated.

Configurations 11 and 12 were defined as $\dot{\gamma}_{cg}$ command configurations with respect to the center of gravity of the aircraft but with the center of rotation located either at the pilot station (34 ft forward of the cg) (Configuration 11) or at the center of gravity (Configuration 12). Both configurations have $\dot{\gamma}$ transfer functions with two zeros at the origin of the $\dot{\gamma}/F_S(s)$ transfer functions, so the denominator contained a $s^2 = 0$ term. The response in γ_{cg} for the configuration in which the center of rotation was located at the pilot station (Configuration 11) was non-minimum phase, with a transfer function zero at $s = +3.86$. A pole was, therefore, placed at this value reflected about the $j\omega$ axis, i.e., at $s = -3.86$. This is the pole value that would minimize the non-minimum phase response effects and is the value that would approximately be achieved if the design were accomplished using linear optimal control methods. Because the $\dot{\gamma}/F_S(s)$ transfer functions were not rational, i.e., the numerator and denominator polynomials had the same number of singularities, an addition pole was added at $s = -2$ to yield a rational response to a pilot command input. The purpose of these configurations was to demonstrate that

1. The $s^2 = 0$ terms, which dominate the long term response, are very detrimental to flying qualities even if the $n_Z(t)$ (and in this case $q(t)$ also) are smooth and well behaved in short term.

2. The response closely resembles the behavior of a system with a transfer function of $\alpha/F_S(s) \doteq K/s$, which is very detrimental to flying qualities.

3. A "command" configuration should be designed independently in the short period and long period dynamics of the vehicle. Although it is believed that very good Level 1 $\dot{\gamma}(t)$ command configurations can be designed, the design should likely be for the short term only, with the long term most effectively designed as an angle of attack command system, i.e., phugoid poles at or near the values of ω_α, ζ_α, zeros.

For Configurations 13 and 14, a third value of $1/\tau_{\theta_2} = 2.0$ was chosen as a large value of $1/\tau_{\theta_2}$ equal to the value of the short period frequency and resulting in a value of CAP that would place the configurations in the Level 1 area for the Category C precision requirement but in the vicinity of the Level 1 - Level 2 boundary with respect to Category A precision requirements. A large value of $1/\tau_{\theta_2}$ was chosen for several reasons:

1. To determine where the lower constant CAP boundary was accurately defined for both approach and for flare and landing.

2. To try to help settle the controversy of whether the constant CAP, ω_n vs n/α requirement of the MIL-F-8785(C) specification or the $\omega_n \tau_{\theta_2}$ vs ζ_{sp} hypothesis of the MIL-F-8785 Handbook and MIL Standard more accurately defines approach, and flare and landing requirements.

3. To test the hypothesis that a pilot might favorably accept a dynamical configuration in which the flight path and attitude responses were in close coordination during the short period response of the vehicle. Not only did the higher $1/\tau_{\theta_2}$ reduce the short period steady state angle of attack change required to maneuver, but also made the pitch rate and angle of attack responses dynamically more similar in the short term.

It was felt that the configurations designed and flown in this program would go a long way toward helping the control system designer properly interpret the MIL-F-8785(C) requirements. Most of the configurations are designed to test the principle that:

1. The short period requirements of MIL-F-8785(C) apply to the angle of attack response of the vehicle.

2. In the long term the system should be designed as an angle of attack command system. The phugoid poles should be placed at or near the low frequency zeros of the $\alpha/F_S(s)$ transfer function.

3. If these objectives are achieved, then the pilot can use the pitch response of the vehicle as an effective surrogate for the flight path response because the vehicle then flies proportional to the direction it is being pointed by the pilot.

ω_{sp} Verification (Configurations 15,16)

Two angle of attack command configurations with different short period frequencies were added as part of the experiment to provide extra data to further refine the time domain criteria. These configurations were similar to Configuration 1 except ω_{sp}= 1 rad/sec for Configuration 15 and ω_{sp} = 3 rad/sec for Configuration 16.

Washout Investigation (Configurations 17-20)

The previous TIFS/Pitch Rate program (Reference 5) briefly investigated the effects of a washout prefilter on a specific pitch rate command configuration (Shuttle Orbiter). It was decided to make a systematic investigation of washouts in this experiment. Configuration 1-2-2 of the Reference 5 experiment was chosen as a baseline Level 2 pitch rate command configuration. This was called Configuration 17 in this program. A washout prefilter was added to the pilot command path with various washout time constants (Configuration 18-20). These configurations differed from all others in that they were mechanized with a q feedback path and a proportional-plus integral in the command path.

Time Delay Control Sensitivity Matrix Configuration Description

The purpose of this part of the test matrix was to determine the effect, if any, of pitch sensitivity on time delay flying qualities effects. The basis of the matrix was Configuration B, a Level 1 conventional angle of attack command system (see Appendix A for step responses).

Three variations in pitch sensitivity were chosen during the calibration test flights. The mid sensitivity was 0.42 deg/sec^2/lb. This value had been found to be a near optimum sensitivity of wheel controllers in previous programs and was verified as such during the calibration flights. Minimum and maximum sensitivity values were chosen, by flight tests, that would still yield Level 1 or borderline Level 1 flying qualities. The minimum value chosen was 0.25 deg/sec^2/lb and the maximum value was 0.63 deg/sec^2/lb. These values were selected by controlling the command gain of the flight control system.

There are a number of ways to change pitch sensitivity, i.e., changing command gain, changing short period frequency, changing short period damping ratio, the addition of prefilters, etc. It was felt then no matter what the method of changing sensitivity, the result is much the same to the pilot. The command gain method of changing sensitivity was chosen as it tended to better isolate the effects of sensitivity while keeping other critical factors constant.

The minimum time delay of Configuration B as implemented in TIFS was 150 ms (measured from the time of wheel force application to maximum slope intercept of the resultant TIFS pitch rate response). Additional transport time delays of 100 ms and 200 ms, respectively, were added to provide three levels of time delay; 150 ms, 250 ms, and 350 ms.

The three levels of sensitivity were used with the three levels of time delay to obtain the matrix of Table 1.

There was also interest in observing the difference, if any, in time delay effects between conventional angle of attack command and pitch rate command flight control systems. A Level 1 pitch rate system, Configuration 17+LL, was chosen as a baseline for an additional 3x3 matrix. During the early part of the program the 350 ms time delay portion of the matrix was flown and the results were so similar to those of the angle of attack command matrix that the balance of this pitch rate command matrix was not flown. Table 1 also shows those three high time delay configurations.

Table 1

TIME DELAY/SENSITIVITY MATRIX

CONFIGURATION NUMBER	FLIGHT CONTROL SYSTEM	SENSITIVITY ($\dot{q}$/lb)	TIME DELAY (ms)
B	B	0.42	150
21	B	0.42	250
22	B	0.42	350
23	B	0.25	150
24	B	0.25	250
25	B	0.25	350
26	B	0.63	150
27	B	0.63	250
28	B	0.63	350
17+LL	17+LL	0.42	150
22A	17+LL	0.42	350
25A	17+LL	0.25	350
28A	17+LL	0.63	350

Throttle Response

The matrix of control effectiveness terms for this experiment was chosen such that a throttle term appeared only in the $\dot{V}$ equation. In other words, it was assumed that the thrust acted through the vehicle center of gravity aligned with the longitudinal axis of the airplane. The responses of the vehicle variables to a throttle command then depended only upon the aerodynamic coupling between the speed change degree of freedom of motion and the other degrees of freedom. The effect of the different flight control system designs is to produce near-decoupling of the speed response from the remaining states for some of the configurations. Configurations 9 and 10, for instance, are completely decoupled dynamically; a throttle input produces only a speed change with no change in attitude or flight path angle.

2.2 DETAILED CONFIGURATION DESCRIPTIONS

2.2.1 Introduction

As previously mentioned, most of the longitudinal characteristics of the configurations were defined by exact pole and zero placement and not by defining an aerodynamic model with various control systems. Once specific poles, zeros, and gains were chosen for each transfer function of a configuration, a phase variable canonical transformation was performed on these transfer functions to convert them to linear, state space matrix-vector representations which defined the longitudinal equations of motion of the aircraft. The configurations which were based on specific aerodynamic configurations were first linearized and then linear F and G matrices were obtained for them.

The Configurations 1-14 were first defined in the frequency domain as transfer functions. It was a relatively easy matter to define the pole-zero cancellation transfer function and the short period frequency constraints of the configurations using this method. The set of transfer functions for a particular configuration were then assembled as a transformation associated with a phase variable canonical form. With the transformation defined, the system of transfer functions were transformed into the time domain for mechanization in the TIFS computer in the familiar state space or matrix-vector format

$$\dot{x}(t) = F\ x(t) + G\ u(t) \tag{1}$$

Equation (1) can be transformed into the phase variable canonical form

$$\dot{y}(t) = A\ y(t) + B\ u(t) \tag{2}$$

using a linear transformation

$$x(t) = T\ y(t) \tag{3}$$

in which the matrices A and B are of the form

$$A = \begin{bmatrix} 0 & 1 & 0 & - & - & - & 0 \\ 0 & 0 & 1 & 0 & - & - & 0 \\ | & & & & & & \\ | & & & & & & \\ 0 & - & - & - & - & 0 & 1 \\ -\alpha_0 & -\alpha_1 & - & - & - & - & \alpha_{n-1} \end{bmatrix} \qquad B = \begin{bmatrix} 0 \\ | \\ | \\ | \\ 0 \\ 1 \end{bmatrix} \tag{4}$$

The coefficients of the matrix A are determined from the characteristic polynomial of the equations of motion

$$|Is-F| = s^n + \alpha_{n-1}s^{n-1} + \ \ldots\ \alpha_1 s + \alpha_0 \tag{5}$$

The transformation matrix T is obtained from the relationship

$$TS = [Is-F]^{adj}G \tag{6}$$

where $[Is-F]^{adj}$ represents the adjugate of the matrix $[Is-F]$ and S is the column matrix $S^T = [1 \ s \ s^2 \ \ldots \ s^{n-1}]$.

A row of the matrix T, therefore, is composed of the coefficients of the numerator polynomial of a transfer function of the system. If the system of Equation (1) is completely controllable and observable, the square transformation matrix T is non-singular and the inverse of T exists. The transformation from the transfer function or phase variable form of Equation (2) and (3) to the state space form of Equation (1) is simply given by

$$\begin{aligned} \dot{x}(t) &= T\,A\,T^{-1}x(t) + T\,B\,u(t) \\ &= F\,x(t) + G\,u(t) \end{aligned} \tag{7}$$

Systems with multicontroller inputs, such as a stick and throttle can be easily accomodated by defining a separate transformation T_i for each controller u_i. The original system of Equation (1) is written as

$$\dot{x}(t) = F\,x(t) + G_1\,u_1(t) + G_2\,u_2(t) + \ \ldots\ G_p\,u_p(t) \tag{8}$$

For each input u_i a separate transformation T_i is defined representing the transfer functions of the states for a particular input u_i. A total of p transformations are defined from the transfer functions of the p inputs. The transformation from these transfer functions to the state space equations is then defined by

$$x(t) = \left(\sum_{i=1}^{p} T\right) A\left(\sum_{i=1}^{p} T\right) \quad x(t) + T_1 B u_1 + T_2 B u_2 \cdots + T_p B u_p \tag{9}$$

Therefore, transfer functions with respect to a throttle command input can be defined independently of those with respect to a stick command. The only requirements (other than controllability and observability) is that the poles of the stick and throttle transfer functions be in common.

Example

A simple example of the use of the phase variable transformation is given below. Assume it is desired to find the augmented equations of motion that would yield the following transfer functions to an elevator and a direct lift surface

$$\frac{q}{\delta_{e_c}}(s) = \frac{M_{\delta_e}(s + 1/\tau_{\theta_2})}{s^2 + 2\zeta\omega_n s + \omega_n^2}$$

$$\frac{\alpha}{\delta_{e_c}}(s) = \frac{M_{\delta_e}}{s^2 + 2\zeta\omega_n s + \omega_n^2}$$

From these transfer functions, the matrices A and T are defined

$$A = \begin{bmatrix} 0 & 1 \\ -\omega_n^2 & -\zeta\omega_n \end{bmatrix} \qquad T = \begin{bmatrix} M_{\delta_e}/\tau_{\theta_2} & M_{\delta_e} \\ M_{\delta_e} & 0 \end{bmatrix}$$

then

$$\begin{bmatrix} \dot{q} \\ \dot{\alpha} \end{bmatrix} = T\,A\,T^{-1}\begin{bmatrix} q \\ \alpha \end{bmatrix} + T\begin{bmatrix} 0 \\ 1 \end{bmatrix} \delta_{e_c}$$

or

$$\begin{bmatrix} \dot{q} \\ \dot{\alpha} \end{bmatrix} = \begin{bmatrix} (1/\tau_{\theta_2} - 2\zeta\omega_n) & (-\omega_n^2 - 1/\tau_{\theta_2}{}^2 + 2\zeta\omega_n/\tau_{\theta_2}) \\ -1 & -1/\tau_{\theta_2} \end{bmatrix} \begin{bmatrix} q \\ \alpha \end{bmatrix} + \begin{bmatrix} M_{\delta_e} \\ 0 \end{bmatrix} \delta_{e_c}$$

$$\begin{bmatrix} \dot{q} \\ \dot{\alpha} \end{bmatrix} = \begin{bmatrix} M_q & M_\alpha \\ -1 & Z_\alpha \end{bmatrix} \begin{bmatrix} q \\ \alpha \end{bmatrix} + \begin{bmatrix} M_{\delta_e} \\ 0 \end{bmatrix} \delta_{e_c}$$

Not only can the transformation described above be used to find the equation of motion equivalent to a set of transfer functions, but as described in Reference 14, a transformation similar to the phase variable transformation be used to define filters or observers. Those observers are then used to define a control law for any aircraft that can place all the poles of a system in accordance with the requirements of MIL-F-8785(C) using a minimum number of sensors and without increasing the order of the closed-loop response to a pilot command or other input to the system.

Though most of the configurations were defined originally by poles and zeros, the general characteristics of all of the configurations were those of a medium transport aircraft. The lateral/directional characteristics were those used in a previous TIFS program (Reference 5). A transport type wheel/column and rudder pedal feel system were also used.

Command gains were chosen during the checkout phase of the program to yield initial pitch sensitivities that were typical of a transport aircraft. It turned out that a nominally good value for the pitch sensitivity yielded a maximum pitch acceleration of approximately 5 deg/sec^2 for a 10 pound step input. Command gains were adjusted on all of the configurations (except those which specifically had sensitivity variations as an experiment variable) to yield this nominal initial sensitivity. It should be noted that for configurations with large pitch rate overshoot the constant initial pitch acceleration resulted in these configurations having higher steady state forces in a pitch maneuver. Steady state pitch forces were eliminated in turns with a system that automatically inserted the proper amount of pitch command to yield the normal acceleration as function of bank angle to hold altitude.

The gain and real root of the angle of attack transfer function were chosen for most of the configurations to put the instantaneous center of rotation at the pilot position, which was 33.8 ft. forward of the center of gravity in the TIFS. This was done to eliminate from the evaluations the lead that one gets from $\dot{q}$ acting through the lever arm distance of the center of rotation to pilot position. The configurations which had different centers of rotation are indicated in the following summary.

2.2.2 Longitudinal Configurations

A brief summary of the specific configurations are now presented, followed by the complete transfer functions and F and G matrices. Table 2 lists a summary of the configuration characteristics.

Configuration B

This was a baseline conventional airplane configuration that was chosen to yield Level I flying qualities, about which time delay and pitch sensitivity variations could be made to investigate their effects on flying qualities. It was based on a TIFS aerodynamic model with increased M_α and M_q derivatives in order to achieve an ω_{sp} = 2 rad/sec and a ζ_{sp} = .7, $1/\tau_{\theta_2}$ = .75 N_z/α = 5.3. The instantaneous center of rotation was 22.2 ft aft of the pilot.

Configurations 1-8

Form basic set to evaluate different command response types.

1. α cmd (short/long term) — $\omega_{sp} = 2$, $\zeta = .7$, $\omega_{ph} = .3$, $1/\tau_{\theta_2} = .5$, $N_z/\alpha = 3.5$
2. q cmd (short/long term) — $\omega_{sp} = 2$, $\zeta = 2.1$, $\omega_\alpha = .3$, $1/\tau_{\theta_2} = .5$, $N_z/\alpha = 3.5$
3. α (short term)/q (long term) — $\omega_{sp} = 2$, $\zeta = .7$, $\omega_\alpha = .3$, $1/\tau_{\theta_2} = .5$, $N_z/\alpha = 3.5$
4. q (short term)/α (long term) — $\omega_{sp} = 2$, $\zeta = 2.1$, $\omega_{ph} = .3$, $1/\tau_{\theta_2} = .5$, $N_z/\alpha = 3.5$
5. α cmd — $\omega_{sp} = 2$, $\zeta = .7$, $\omega_{ph} = .1$, $1/\tau_{\theta_2} = .9$, $N_z/\alpha = 6.3$

Table 2
CONFIGURATION CHARACTERISTICS

CONFIGURATION	ω_{sp} (r/s)	ζ_{sp}	ω_{ph} (r/s)	ζ_{ph}	$1/\tau_{\theta_2}$ (r/s)	N_z/α (g's/rad)
B	2	.7	.16	.095	.75	5.3
1	2	.7	.3	.1	.5	3.5
2	2	2.1	(0)	(-.1)	.5	3.5
3	2	.7	(0)	(-.1)	.5	3.5
4	2	1.3	.3	.1	.5	3.5
5	2	.7	.1	.1	.9	6.3
6	2	1.3	(0)	(-.1)	.9	6.3
7	2	.7	(0)	(-.1)	.9	6.3
8	2	1.3	.1	.1	.9	6.3
9	2	.7	(0)	(-.1)	.5	3.5
10	2	1.3	(0)	(-.1)	.5	3.5
11	2	.7	(0)	(0)	.5	3.5
12	2	.7	(0)	(0)	.5	3.5
13	2	.7	.3	.1	2	14.
14	2	1.3	(0)	(-.1)	2	14.
15	1	.7	.3	.1	.35	2.5
16	3	.7	.3	.1	1	7
17 thru 20	2.9	.78	(0)	(-.05)	.72	5
21 thru 28	2	.7	.16	.095	.75	5.3
17 + Lead/Lag & 21A thru 28A	2.9	.78	(0)	(-.05)	.72	5

Note: Configurations with phugoid characteristics listed as (-) have real roots where indicated.

$1/\tau_{\theta_1} = 0.1$ for Configurations 1-14

6. q cmd $\omega_{sp} = 2,\ \zeta = 1.3,\ \omega_\alpha = .1,\ 1/\tau_{\theta_2} = .9,$
$N_z/\alpha = 6.3$

7. α/q cmd $\omega_{sp} = 2,\ \zeta = .7,\ \omega_\alpha = .1,\ 1/\tau_{\theta_2} = .9,$
$N_z/\alpha = 6.3$

8. q/α cmd $\omega_{sp} = 2,\ \zeta = 1.3,\ \omega_{ph} = .1,\ 1/\tau_{\theta_2} = .9,$
$N_z/\alpha = 6.3$

$1/\tau_{\theta_1} = 0.1$ for all configurations.

Configurations 9, 10

The low frequency (phugoid) response is eliminated in both q and α by proper zero locations based on Configurations 1, 2.

9. α cmd - low frequency zeros of q and α at $(-1/\tau_{\theta_1}, 0) = (.1, 0)$,
$1/\tau_{\theta_2} = .5,\ N_z/\alpha = 3.5$

10. q cmd "

Configurations 11, 12

Investigate $\dot{\gamma}$ (or N_z) command systems referenced to different locations, all with $\omega_{sp} = 2,\ \zeta = .7,\ 1/\tau_{\theta_2} = .5,\ N_z/\alpha = 3.5$.

11. $\dot{\gamma}$ cmd w/r/t CG - CR at pilot
12. $\dot{\gamma}$ cmd w/r/t CG - CR at CG 33.8 ft aft of pilot

Configurations 13, 14

Investigate the use of direct lift control to increase effective $1/\tau_{\theta_2}$.

13. α cmd hi $1/\tau_{\theta_2} = 2,\ \omega_{sp} = 2,\ \zeta = .7,\ N_z/\alpha = 14$
14. q cmd hi $1/\tau_{\theta_2} = 2,\ \omega_{sp} = 2,\ \zeta = 1,\ N_z/\alpha = 14$

Configurations 15, 16

Short period frequency variations to gather data for flying qualities time history criteria refinement.

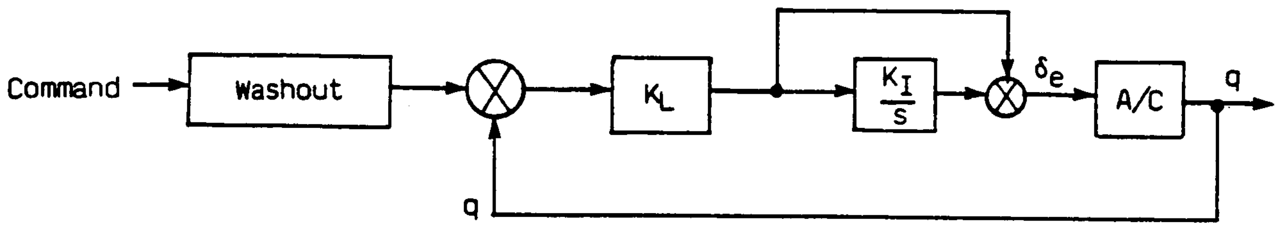
15. α cmd $\omega_{sp} = 1,\ \zeta = .7,\ \omega_{ph} = .3,\ 1/\tau_{\theta_2} = .35,\ N_z/\alpha = 2.5$
(Was not flown and will not be discussed further)

16. α cmd $\omega_{sp} = 3,\ \zeta = .7,\ \omega_{ph} = .3,\ 1/\tau_{\theta_2} = 1.0,\ N_z/\alpha = 7.0$

Configurations 17-20

Investigate the effect of various washout prefilters on a Level 2 configuration from the previous pitch rate program (Reference 5, Configuration 1-2-2). Instantaneous center of rotation was 14 ft aft of the pilot.

These configurations differ from all the others in that they are mechanized with a q-feedback path and a proportional plus integral in the command path:



$K_L = 4$

$K_I = 2$

Washout = $(s)/(s + 1/\tau_{WO})$

17. (1-2-2) $\omega_{sp} = 2.8$, $1/\tau_{\theta_2} = .7$, $N_Z/\alpha = 5$ No washout (W.O.)
18. W.O. $1/\tau_{WO} = .05$
19. W.O. $1/\tau_{WO} = .10$
20. W.O. $1/\tau_{WO} = .20$

Configurations 21-28

Time delay and command gain (sensitivity) variations on a baseline Level 1 conventional airplane configuration to gather data for criteria refinement.

21.	Configuration B -	nominal sensitivity with	100 ms	extra delay
22.	"	"	200 ms	"
23.	"	.6(Nominal) sensitivity	0 ms	"
24.	"	"	100 ms	"
25.	"	"	200 ms	"
26.	"	1.5(Nominal) sensitivity	0 ms	"
27.	"	"	100 ms	"
28.	"	"	200 ms	"

The time delay/sensitivity configuration matrix is shown below:

	Extra time delay (ms)		
Sensitivity	0	+100	+200
.6(Nominal)	23	24	25
Nominal	B	21	22
1.5(Nominal)	26	27	28

The baseline configuration had an effective time delay of 150 ms so the increased time delay configurations had effective time delays of 250 and 350 ms. The three values of sensitivity were defined during checkout phase of the program and were chosen to span a range of sensitivities which would still yield Level I flying qualities with the minimum time delay value.

Four additional configurations were added during the evaluation phase of the program to gather time delay/sensitivity data on a pitch rate command type airplane. The baseline configuration chosen for this set was Configuration 17 with a lead/lag filter. This was Configuration 4-2-2 of the previous TIFS/Pitch Rate program. The block diagram of this configuration is the same as that shown above for Configuration 17 with a lead/lag filter: (1.22 s + 1)/(.5 s + 1) replacing the washout block. The nominal command gain had to be reduced by a factor of .5/1.22 to keep the initial sensitivity the same.

The time delay/sensitivity configuration matrix based on Configuration 17 + Lead/Lag is:

	Extra time delay (ms)		
Sensitivity	0	+100	+200
.6(Nominal)	23A	24A	25A
Nominal	17 + L/L	21A	22A
1.5(Nominal)	26A	27A	28A

Only Configuration 17 + Lead/Lag, 22A, 25A, and 28A were flown.

2.2.2.1 Transfer Functions

This section presents a tabulation of the transfer functions of each configuration. It is written in the shorthand notation where:

$$K(a)[\zeta, \omega] \text{ is equivalent to } K(s + a)[s^2 + 2\zeta\omega s + \omega^2]$$

The transfer functions are all with respect to force input (F_{ES}). The following factors are found in each configuration and are not listed in the individual transfer functions:

Feel system:	$20^2/[.7, 20]$
Gradient:	1/12 in/lb
Actuator:	$20^2/[.7, 20]$

The command gain, K_c (deg/inch), is listed separately for each configuration and should multiply the gain of each numerator. The command gain was chosen to yield the same maximum $\dot{q}$ for each configuration for a step input (see time histories in Appendix A). A pure transport time delay of 60 ms should also be added to each transfer function to take into account model following lags in the TIFS. This 60 ms delay has been added in the time histories shown in Appendix A.

The velocity numerator to stick command input is the same for all configurations (except for B, 17-20, 17 + Lead/Lag, 21-28 and 21A-28A).

$$N_V = -.05(1)(-15)$$

<u>Configuration B</u> Baseline conventional airplane $\omega_{sp} = 2,$ $\zeta_{sp} = .7,$ $1/\tau_{\theta_2} = .75,$ $\omega_{ph} = .16$

$K_C = -3.3$
$N_q = 19.5(0)(.069)(.75)$
$N_\alpha = 1.(21.1)[.09, .19]$
$N_V = -.33\ (1.68)(5.03)(-3.10)$

$D = [.7, 2][.095, .16]$

Configuration 1 α-cmd, $\omega_{sp} = 2$, $\zeta_{sp} = .7$, $1/\tau_{\theta_2} = .5$, $\omega_{ph} = .3$

$K_c = -7.8$
$N_q = -1(0)(.1)(.5)$
$N_\alpha = -.15(3.7)[.1, .3]$

$D = [.7, 2][.1, .3]$

Configuration 2   q-cmd, $\omega_{sp} = 2$, $\zeta_{sp} = 2.1$, $1/\tau_{\theta_2} = .5$

$K_c = -11.7$
$N_q = -1(0)(.1)(.5)$
$N_\alpha = -.15(3.7)[.1, .3]$

$D = (0)(.1)(.5)(8)$

Configuration 3 α/q, $\omega_{sp} = 2$, $\zeta_{sp} = .7$, $1/\tau_{\theta_2} = .5$

$K_c = -7.8$
$N_q = -1(0)(.1)(.5)$
$N_\alpha = -.15(3.7)[.1, .3]$

$D = (0)(.1)[.7, 2]$

Configuration 4 q/α, $\omega_{sp} = 2$, $\zeta_{sp} = 2.1$, $1/\tau_{\theta_2} = .5$, $\omega_{ph} = .3$

$K_c = -11.7$
$N_q = -1(0)(.1)(.5)$
$N_\alpha = -.15(3.7)[.1, .3]$

$D = (.5)(8)[.1, .3]$

Configuration 5 α-cmd, $\omega_{sp} = 2$, $\zeta_{sp} = .7$, $1/\tau_{\theta_2} = .9$, $\omega_{ph} = .1$

$K_c = -7.8$
$N_q = -1(0)(.1)(.9)$
$N_\alpha = -.15(3.7)[.1, .1]$

$D = [.7, 2][.1, .1]$

Configuration 6 q-cmd, $\omega_{sp} = 2$, $\zeta_{sp} = 1.3$, $1/\tau_{\theta_2} = .9$

$K_c = -9.1$

$N_q = -1(0)(.1)(.9)$

$N_\alpha = -.15(3.7)[.1, .1]$

$D = (0)(.1)(.9)(4.4)$

Configuration 7 α/q, $\omega_{sp} = 2$, $\zeta_{sp} = .7$, $1/\tau_{\theta_2} = .9$

$K_c = -7.8$

$N_q = -1(0)(.1)(.9)$

$N_\alpha = -.15(3.7)[.1, .1]$

$D = (0)(.1)[.7, 2]$

Configuration 8 q/α, $\omega_{sp} = 2$, $\zeta = 1.3$, $1/\tau_{\theta_2} = .9$, $\omega_{ph} = .1$

$K_c = -9.1$

$N_q = -1(0)(.1)(.9)$

$N_\alpha = -.15(3.7)[.1, .1]$

$D = (.9)(4.4)[.1, .1]$

Configuration 9 α, q decoupled from phugoid α-cmd, $\omega_{sp} = 2$, $1/\tau_{\theta_2} = .5$

$K_c = -7.8$

$N_q = -1(0)(.1)(.5)$

$N_\alpha = -.15(0)(.1)(3.7)$

$D = (0)(.1)[.7, 2]$

Configuration 10 α, q decoupled from phugoid, q-cmd, $\omega_{sp} = 2$, $1/\tau_{\theta_2} = .5$

$K_c = -11.7$

$N_q = -1(0)(.1)(.5)$

$N_\alpha = -.15(0)(.1)(3.7)$

$D = (0)(.1)(.5)(8)$

<u>Configuration 11</u> $\dot{\gamma}$-cmd pilot at CR, 34' fwd of CG, $\omega_{sp} = 2$, $1/\tau_{\theta_2} = .5$

$K_c = -40$

$N_q = -1(0)(.1)(.5)$

$N_\alpha = -.15(3.7)[.1, .3]$

$N_{n_{z_{CG}}} = -.018(0)(0)(.955)(-3.86)$

$D = (0)(0)(.955)(3.86)(2)$

<u>Configuration 12</u> $\dot{\gamma}$-cmd pilot 34' fwd of CR, CR at CG, $\omega_{sp} = 2$, $1/\tau_{\theta_2} = .5$

$K_c = -25.48$

$N_q = -1.(0)(.1)(.5)$

$N_\alpha = -.555\ [.1, .3]$

$N_{n_{z_{CG}}} = -.055(0)(0)(1.274)$

$D = (0)(0)(1.274)[.7, 2]$

<u>Configuration 13</u> High $1/\tau_{\theta_2} = 2$, α-cmd, $\omega_{sp} = 2$, $\zeta_{sp} = .7$

$K_c = -7.8$

$N_q = -1(0)(.1)(2)$

$N_\alpha = -.15(3.7)[.1, .3]$

$D = [.7, 2][.1, .3]$

<u>Configuration 14</u> High $1/\tau_{\theta_2} = 2$, q-cmd, $\omega_{sp} = 2$, $\zeta_{sp} = 1$

$K_c = -7.8$

$N_q = -1(0)(.1)(2)$

$N_\alpha = -.15(3.7)[.1, .3]$

$D = (0)(.1)(2)(2)$

Configuration 15 Not flown

Configuration 16 α-cmd, $\omega_{sp} = 3$, $\zeta_{sp} = .7$, $1/\tau_{\theta_2} = 1$, $\omega_{ph} = .3$

$K_C = -7.8$

$N_q = -1(0)(.1)(1)$

$N_\alpha = -.15(3.7)[.1, .3]$

$D = [.7, 3][.1, .3]$

Configuration 17 Configuration 1-2-2 of previous TIFS/Pitch Rate program with no washout

$K_C = -1.7$

$N_q = -1417(0)(.056)(.72)(2)$

$N_\alpha = -129.5(11.4)(2)[.106, .2]$

$N_V = -12.3(.878)(2)(-54)$

$D = (0)(.052)(.82)[.78, 2.9][.7, 17.1]$

NOTE: [.7, 17.1] is the actuator pole which has migrated from [.7, 20] as it is now in the loop.

Configuration 17 + Lead/Lag Configuration 4-2-2 of previous TIFS/Pitch Rate program where lead/lag is:

$$(1.22\ s + 1)/(.5\ s + 1) = 2.44\ (s + .82)/(s + 2)$$

$$\left.\begin{array}{ll} K_C = & -.74 \\ N_q,\ N_\alpha,\ N_V - & \text{same as Configuration 17 with 2.44 (.82) factors} \\ D = & \text{same as Configuration 17 with (2.) factor} \end{array}\right\} \frac{2.44\ (s + .82)}{(s + 2)}$$

Configuration 18 Configuration 17 with $1/\tau_{WO} = .05$

$K_C = -1.7$

N_q, N_α, N_V - same as Configuration 17 with (0) factor

D = same as Configuration 17 with (.05) factor

$$\left.\right\} \frac{s}{s+.05}$$

Configuration 19 Configuration 17 with $1/\tau_{WO} = .1$

$K_C = -1.7$

N_q, N_α, N_V - same as Configuration 17 with (0) factor

D = same as Configuration 17 with (.1) factor

$$\left.\right\} \frac{s}{s+.1}$$

Configuration 20 Configuration 17 with $1/\tau_{WO} = .2$

$K_C = -1.7$

N_q, N_α, N_V - same as Configuration 17 with (0) factor

D = same as Configuration 17 with (.2) factor

$$\left.\right\} \frac{s}{s+.2}$$

Configurations 21-28 Time Delay/Sensitivity variations based on a conventional airplane

These configurations are identical to the previously described Configurations B except for the command gain, K_C and extra time delay above the baseline 150 ms.

Config 21 = Config B with $K_C = -3.3$, +100 ms delay
Config 22 = Config B with $K_C = -3.3$, +200 ms delay
Config 23 = Config B with $K_C = -2.0$, +0 ms delay
Config 24 = Config B with $K_C = -2.0$, +100 ms delay
Config 25 = Config B with $K_C = -2.0$, +200 ms delay
Config 26 = Config B with $K_C = -5.0$, + 0 ms
Config 27 = Config B with $K_C = -5.0$, +100 ms delay
Config 28 = Config B with $K_C = -5.0$, +200 ms delay

Configurations 21A-28A   Time Delay, Sensitivity variations based on the pitch rate command system.

These configurations are identical to the previously described Configuration 17 + Lead/Lag except for the command gain, Kc, and extra time delay above the baseline 150 ms.

Config 21A = Config 17 + L/L with K_C = -.74, +100 ms
Config 22A = Config 17 + L/L with K_C = -.74, +200 ms
Config 23A = Config 17 + L/L with K_C = -.44, +0 ms
Config 24A = Config 17 + L/L with K_C = -.44, +100 ms
Config 25A = Config 17 + L/L with K_C = -.44, +200 ms
Config 26A = Config 17 + L/L with K_C = -1.11, +0 ms
Config 27A = Config 17 + L/L with K_C = -1.11, +100 ms
Config 28A = Config 17 + L/L with K_C = -1.11, +200 ms

Only Configurations 22A, 25A, and 28A were flown.

2.2.2.2 Matrix-Vector Representation

This section presents the linearized model equations of motion in the form of:

$$\dot{X} = Fx + Gu$$

where the state vector $X = \begin{bmatrix} q\text{, deg/sec} \\ \theta\text{, deg} \\ \alpha\text{, deg} \\ V\text{, ft/sec} \end{bmatrix}$

the control vector $u = [\delta_e\text{, deg}]$

In addition, each configuration has a second order feel system (ζ = .7, ω = 20 rad/sec) and a second order actuator (ζ = .7, ω = 20 rad/sec) and a command gain K_C (deg/inch). The feel system gradient is 12 lb/in for all configurations. There is also an additional 60 ms of pure transport time delay to take into account model following lags in the TIFS.

The F and G matrices for each of the configurations are presented:

$$\begin{bmatrix} \dot{q} \\ \dot{\theta} \\ \dot{\alpha} \\ \dot{V} \end{bmatrix} = F \begin{bmatrix} q \\ \theta \\ \alpha \\ V \end{bmatrix} + G \quad \delta_e$$

<u>Configuration B</u> Baseline conventional airplane, $\omega_{sp} = 2$, $1/\tau_{\theta_2} = .75$

$$F = \begin{bmatrix} -1.9000 & .0000 & -2.2900 & .0246 \\ 1.0000 & .0000 & .0000 & .0000 \\ 1.0000 & .0000 & -.9010 & -.0711 \\ .0000 & -.5614 & .2618 & -.0382 \end{bmatrix} \quad G = \begin{bmatrix} 1.9500 \\ .0000 \\ -.1010 \\ -.0326 \end{bmatrix} \quad K_c = -3.3$$

<u>Configuration 1</u> α-cmd $\omega_{sp} = 2$ $1/\tau_{\theta_2} = .5$

$$F = \begin{bmatrix} -1.4047 & -.1536 & -5.7022 & .0896 \\ 1.0000 & .0000 & .0000 & .0000 \\ .3528 & .0165 & -1.4520 & -.0237 \\ -.1440 & -1.3427 & 1.2930 & -.0033 \end{bmatrix} \quad G = \begin{bmatrix} -1.0000 \\ .0000 \\ -.1500 \\ .0000 \end{bmatrix} \quad K_c = -7.8$$

<u>Configuration 2</u> q-cmd $\omega_{sp} = 2$ $1/\tau_{\theta_2} = .5$

$$F = \begin{bmatrix} -8.0000 & .0000 & .0000 & .0000 \\ 1.0000 & .0000 & .0000 & .0000 \\ -.6365 & .0396 & -.5967 & -.0371 \\ -.1440 & -1.3427 & 1.2930 & -.0033 \end{bmatrix} \quad G = \begin{bmatrix} -1.0000 \\ .0000 \\ -.1500 \\ .0000 \end{bmatrix} \quad K_c = -11.7$$

<u>Configuration 3</u> α/q $\omega_{sp} = 2$ $1/\tau_{\theta_2} = .5$

$$F = \begin{bmatrix} -1.5868 & -.5784 & -4.7550 & -.3556 \\ 1.0000 & .0000 & .0000 & .0000 \\ .3255 & -.0472 & -1.3099 & -.0905 \\ -.1440 & -1.3427 & 1.2930 & -.0033 \end{bmatrix} \quad G = \begin{bmatrix} -1.0000 \\ .0000 \\ -.1500 \\ .0000 \end{bmatrix} \quad K_c = -7.8$$

Configuration 4 q/α $\omega_{sp} = 2$ $1/\tau_{\theta_2} = .5$

$$F = \begin{bmatrix} -8.1720 & -.4709 & 1.4131 & .5428 \\ 1.0000 & .0000 & .0000 & .0000 \\ -.6623 & -.0311 & -.3847 & .0443 \\ -.1440 & -1.3427 & 1.2930 & -.0033 \end{bmatrix} \quad G = \begin{bmatrix} -1.0000 \\ .0000 \\ -.1500 \\ .0000 \end{bmatrix} \quad K_C = -11.7$$

Configuration 5 α-cmd $\omega_{sp} = 2$ $1/\tau_{\theta_2} = .9$

$$F = \begin{bmatrix} -1.0030 & .0652 & -5.4467 & .0205 \\ 1.0000 & .0000 & .0000 & .0000 \\ .3930 & -.0302 & -1.7197 & -.0085 \\ .0216 & -.8224 & .1894 & -.0973 \end{bmatrix} \quad G = \begin{bmatrix} -1.0000 \\ .0000 \\ -.1500 \\ .0000 \end{bmatrix} \quad K_C = -7.8$$

Configuration 6 q-cmd $\omega_{sp} = 2$ $1/\tau_{\theta_2} = .9$

$$F = \begin{bmatrix} -4.4000 & .0000 & .0000 & .0000 \\ 1.0000 & .0000 & .0000 & .0000 \\ -.1166 & -.0400 & -.9027 & -.0115 \\ .0216 & -.8224 & .1894 & -.0973 \end{bmatrix} \quad G = \begin{bmatrix} -1.0000 \\ .0000 \\ -.1500 \\ .0000 \end{bmatrix} \quad K_C = -9.1$$

Configuration 7 α/q $\omega_{sp} = 2$ $1/\tau_{\theta_2} = .9$

$$F = \begin{bmatrix} -1.1634 & -.2828 & -4.9108 & -.0706 \\ 1.0000 & .0000 & .0000 & .0000 \\ .3689 & -.0824 & -1.6393 & -.0221 \\ .0216 & -.8224 & .1894 & -.0973 \end{bmatrix} \quad G = \begin{bmatrix} -1.0000 \\ .0000 \\ -.1500 \\ .0000 \end{bmatrix} \quad K_C = -7.8$$

Configuration 8 q/α $\omega_{sp} = 2$ $1/\tau_{\theta_2} = .9$

$$F = \begin{bmatrix} -4.3232 & .3206 & .0216 & .0914 \\ 1.0000 & .0000 & .0000 & .0000 \\ -.1051 & .0081 & -.8995 & .0022 \\ .0216 & -.8224 & .1894 & -.0973 \end{bmatrix} \quad G = \begin{bmatrix} -1.0000 \\ .0000 \\ -.1500 \\ .0000 \end{bmatrix} \quad K_C = -11.7$$

<u>Configuration 9</u> α-cmd uncoupled from phugoid

$$F = \begin{bmatrix} -1.4094 & .0000 & -5.9376 & .0000 \\ 1.0000 & .0000 & .0000 & .0000 \\ .3436 & .0000 & -1.3906 & .0000 \\ -.1922 & -1.5000 & 1.6146 & -.1000 \end{bmatrix} \quad G = \begin{bmatrix} -1.0000 \\ .0000 \\ -.1500 \\ .0000 \end{bmatrix} \quad K_C = -7.8$$

<u>Configuration 10</u> q-cmd uncoupled from phugoid

$$F = \begin{bmatrix} -8.0000 & .0000 & .0000 & .0000 \\ 1.0000 & .0000 & .0000 & .0000 \\ -.6450 & .0000 & -.5000 & .0000 \\ -.1922 & -1.5000 & 1.6146 & -.1000 \end{bmatrix} \quad G = \begin{bmatrix} -1.0000 \\ .0000 \\ -.1500 \\ .0000 \end{bmatrix} \quad K_C = -11.7$$

<u>Configuration 11</u> $\dot{\gamma}$-cmd CR at Pilot 34' fwd of CG

$$F = \begin{bmatrix} -3.7598 & .3405 & -3.0348 & -.1796 \\ 1.0000 & .0000 & .0000 & .0000 \\ -.0005 & .0907 & -1.0519 & -.0641 \\ -.1440 & -1.3427 & 1.2930 & -.0033 \end{bmatrix} \quad G = \begin{bmatrix} -1.0000 \\ .0000 \\ -.1500 \\ .0000 \end{bmatrix} \quad K_C = -20$$

also requires 2/(s + 2) prefilter

<u>Configuration 12</u> $\dot{\gamma}$-cmd CR at CG 34' aft of pilot

$$F = \begin{bmatrix} -2.2000 & .0815 & -4.8569 & -.3184 \\ 1.0000 & .0000 & .0000 & .0000 \\ .5550 & .0261 & -.5837 & -.0372 \\ .0500 & -1.3336 & 1.0891 & -.0163 \end{bmatrix} \quad G = \begin{bmatrix} -1.0000 \\ .0000 \\ .0000 \\ .0000 \end{bmatrix} \quad K_C = -20$$

also requires 1.274/(s + 1.274) prefilter

Configuration 13 α-cmd high $1/\tau_{\theta_2} = 2$

$$F = \begin{bmatrix} .6294 & -.1467 & -9.2630 & -.1766 \\ 1.0000 & .0000 & .0000 & .0000 \\ .6266 & -.0514 & -3.2772 & -.1602 \\ .2751 & -.4206 & -1.5010 & -.2122 \end{bmatrix} \quad G = \begin{bmatrix} -1.0000 \\ .0000 \\ -.1500 \\ .0000 \end{bmatrix} \quad K_c = -7.8$$

Configuration 14 q-cmd high $1/\tau_{\theta_2} = 2$

$$F = \begin{bmatrix} -2.0000 & .0000 & .0000 & .0000 \\ 1.0000 & .0000 & .0000 & .0000 \\ .2322 & -.0294 & -1.8878 & -.1337 \\ .2761 & -.4206 & -1.5010 & -.2122 \end{bmatrix} \quad G = \begin{bmatrix} -1.0000 \\ .0000 \\ -.1500 \\ .0000 \end{bmatrix} \quad K_c = -7.8$$

Configuration 15 α-cmd $\omega_{sp} = 1$ $1/\tau_{\theta_2} = .35$

$$F = \begin{bmatrix} -.8546 & -.1724 & -1.0357 & .0429 \\ 1.0000 & .0000 & .0000 & .0000 \\ .4445 & .0461 & -.6635 & -.0241 \\ -.2672 & -1.7754 & 2.1148 & .0581 \end{bmatrix} \quad G = \begin{bmatrix} -1.0000 \\ .0000 \\ -.1500 \\ .0000 \end{bmatrix} \quad K_c = -7.8$$

Configuration 16 α-cmd $\omega_{sp} = 3$ $1/\tau_{\theta_2} = 1.0$

$$F = \begin{bmatrix} -1.0121 & -.0585 & -14.3193 & .1176 \\ 1.0000 & .0000 & .0000 & .0000 \\ .3972 & -.0135 & -3.1479 & -.0479 \\ .0500 & -.7500 & .0000 & -.1000 \end{bmatrix} \quad G = \begin{bmatrix} -1.0000 \\ .0000 \\ -.1500 \\ .0000 \end{bmatrix} \quad K_c = -7.8$$

Configurations 17-20

Based on previous TIFS/Pitch Rate Configuration 1-2-2: It has loop gain $K_L = 4$, integral gain $K_I = 2$ (see block diagram in previous section) Unaugmented F and G:

$$F = \begin{bmatrix} -.4880 & .0000 & .5200 & .0053 \\ 1.0000 & .0000 & .0000 & .0000 \\ 1.0000 & .0000 & -.6870 & -.0740 \\ .0000 & -.5610 & .0950 & -.0460 \end{bmatrix} \quad G = \begin{bmatrix} -.8860 \\ .0000 \\ -.0810 \\ .0000 \end{bmatrix}$$

Configuration 17 with no washout $K_c = -1.7$

Configuration 17 + Lead/Lag Same as Configuration 4-2-2 of previous TIFS/ Pitch Rate program, uses F and G of 17

requires (1.22 s + 1) (.5 s + 1) prefilter $K_c = -.74$

Configuration 18 Configuration 17 + washout $1/\tau_{wo} = .05$

requires s/(s + .05) prefilter $K_c = -1.7$

Configuration 19 Configuration 17 + washout $1/\tau_{wo} = .10$

requires s/(s + .10) prefilter $K_c = -1.7$

Configuration 20 Configuration 17 + washout $1/\tau_{wo} = .20$

requires s/(s + .20) prefilter $K_c = -1.7$

Configurations 20-28 Time delay/sensitivity variations based on conventional Configuration B. F and G matrices are the same as those for Configuration B.

Configuration 21 = Configuration B with $K_c = -3.3$, +100 ms

Configuration 22 = Configuration B with $K_c = -3.3$, +200 ms

Configuration 23 = Configuration B with $K_c = -2.0$, +0 ms

Configuration 24 = Configuration B iwth $K_c = -2.0$, +100 ms

Configuration 25 = Configuration B with $K_c = -2.0$, +200 ms

Configuration 26 = Configuration B with $K_c = -5.0$ +0 ms

Configuration 27 = Configuration B with $K_c = -5.0$ +100 ms

Configuration 28 = Configuration B with $K_c = -5.0$ +200 ms

Configurations 21A-28A — Time delay/Sensitivity variations based on pitch rate command Configuration 17 + Lead/Lag. F and G matrices are the same as those for Configuration 17 + Lead/Lag.

Configuration 21A = Configuration 17 + L/L with K_C = -.74, +100 ms

Configuration 22A = Configuration 17 + L/L with K_C = -.74, +200 ms

Configuration 23A = Configuration 17 + L/L with K_C = -.44, +0 ms

Configuration 24A = Configuration 17 + L/L with K_C = -.44, +100 ms

Configuration 25A = Configuration 17 + L/L with K_C = -.44, +200 ms

Configuration 26A = Configuration 17 + L/L with K_C = -1.11, +0 ms

Configuration 27A = Configuration 17 + L/L with K_C = -1.11, +100 ms

Configuration 28A = Configuration 17 + L/L with K_C = -1.11, +200 ms

Time histories for a 10 pound F_{ES} step (5 sec in, 5 sec out) are shown in Appendix A. Also presented are the first one second of the time histories on an expanded scale, so the effective time delay in the responses can be seen. The effective time delay in the pitch rate response (maximum slope intercept method) is shown in Table 3.

Table 3

MEASURED EFFECTIVE TIME DELAY

(PITCH RATE MAXIMUM SLOPE INTERCEPT)

(Includes All Systems Delay and Model Following Lags, 60 ms)

Configuration	Time Delay (ms)
1	160
2	150
3	160
4	140
5	160
6	150
7	160
8	150
9	160
10	150
11	250
12	290
13	160
14	160
16	160
17	150
17 + Lead/Lag, 23A, 26A	150
18	150
19	150
20	150
B, 23, 26	150
21, 24, 27	250
22, 25, 28	350
21A, 24A, 27A	250
22A, 25A, 28A	350

2.2.3 Lateral/Directional Description

The lateral/directional aerodynamics and control system were the same as those used for the previous TIFS/Pitch Rate program. The aircraft model had the following characteristics.

Constant Characteristics

Weight (W)	=	193,000 lbs.
Mass (M)	=	5999.4 slugs
Wing Area (S)	=	2147 ft^2
Wing Span (b)	=	157 ft
Wing Chord (c)	=	15.074 ft
I_{xx}	=	4,003,900 slug-ft^3
I_{yy}	=	5,408,550 slug-ft^3
I_{zz}	=	9,184,470 slug-ft^3
I_{xy}	=	223,410 slug-ft^3

Trim Conditions

V_{Trim}	=	132 KIAS (223 fps)
$\bar{q}$	=	59.14 psf

Lateral/Directional Non-Dimensional Derivatives (per degree)

C_{y_β}	-0.03136
C_{y_p}	0.00563
C_{y_r}	0.01345
$C_{y_{\delta_r}}$	0.00536
C_{ℓ_β}	-0.00256
C_{ℓ_p}	-0.01022
C_{ℓ_r}	0.00749
$C_{L_{\delta_a}}$	0.00148
$C_{\ell_{\delta_{SP}}}$	0.00023

Lateral/Directional Non-Dimensional Derivatives (per degree) (Cont'd)

$C_{\ell_{\delta_r}}$	0.00050
C_{n_β}	0.00394
C_{n_p}	-0.00074
C_{n_r}	-0.00552
$C_{n_{\delta_a}}$	0.00023
$C_{n_{\delta_{SP}}}$	0.00024
$C_{n_{\delta_r}}$	-0.00169

The lateral/directional flight control system is shown in Figures 5 and 6. The gains were adjusted to achieve the following characteristics, which were found to be Level 1 and consequently "transparent" to the longitudinal investigation.

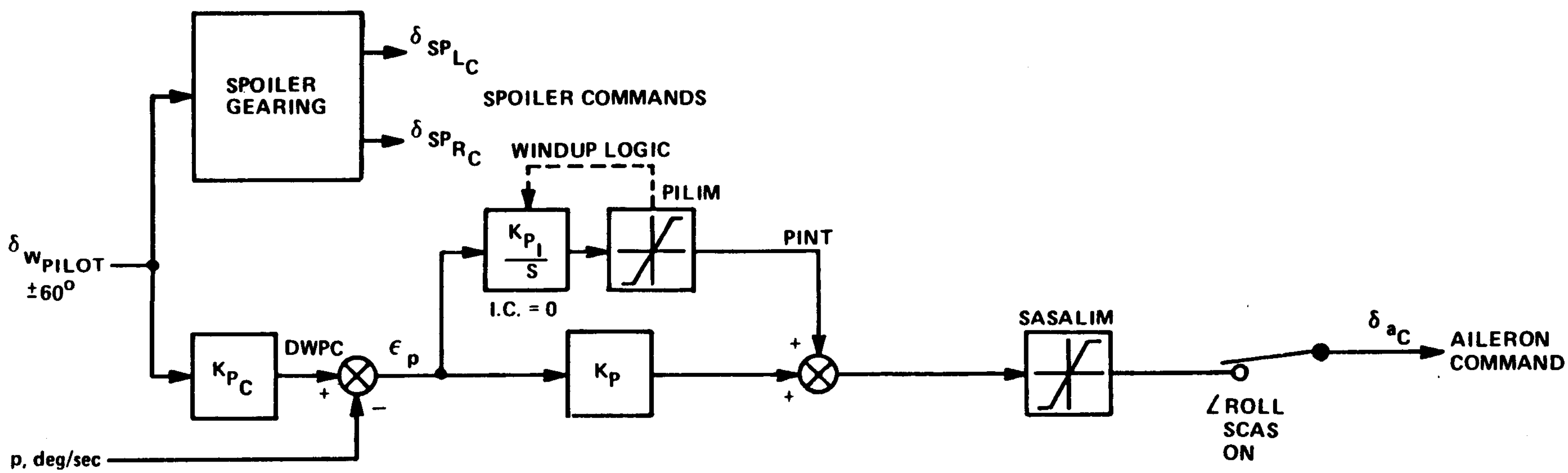
Dutch Roll Mode

ω_n (rad/sec)	0.768
ζ	0.297
$\zeta\omega_n$ (rad/sec)	0.228
P (sec)	8.57
ϕ/β	0.188

Roll Spiral Mode

τ_r (sec)	-
τ_s (sec)	-
ω_{rs} (rad/sec)	4.741
ζ_{rs}	0.369
$\zeta_{rs}\ \omega_{rs}$ (rad/sec)	1.748
P_{rs} (sec)	1.47

Effective roll mode time constant
(time to 63% max roll rate) 1.0 sec

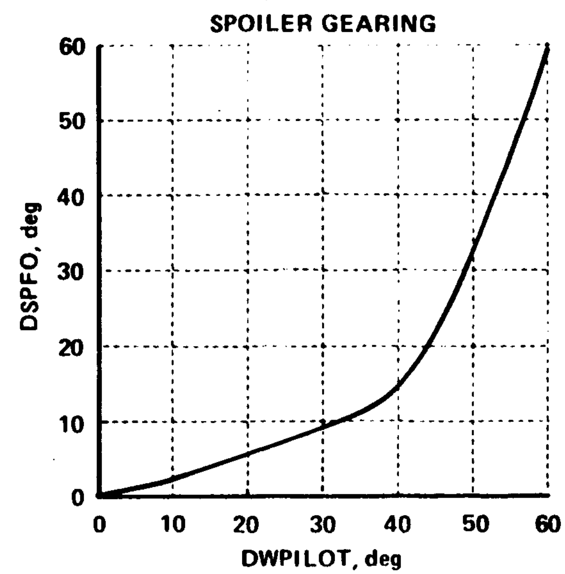


$$DW = \delta_{W_{PILOT}} * \frac{113}{60}$$

$$\delta_{SP_{L_C}} = \begin{cases} 0., & \delta_{W_{PILOT}} > 0. \\ DSPFO, & \delta_{W_{PILOT}} \leqslant 0. \end{cases}$$

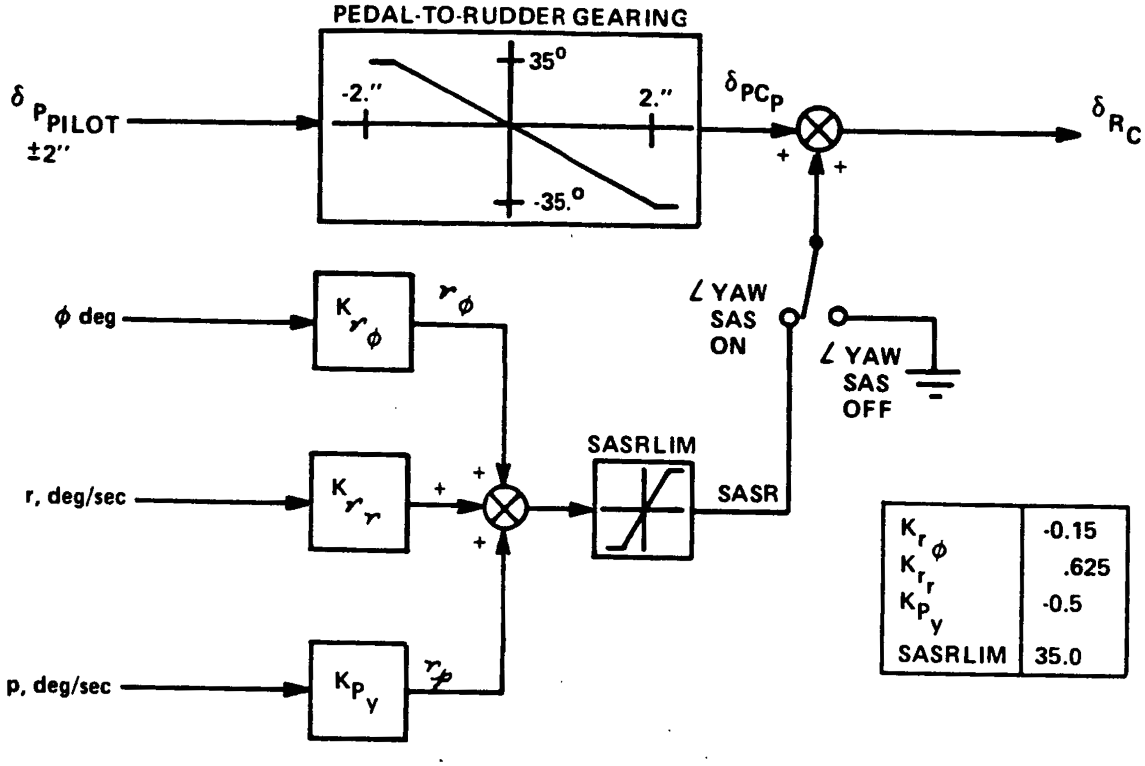
$$\delta_{SP_{R_C}} = \begin{cases} DSPFO, & \delta_{W_{PILOT}} > 0. \\ 0., & \delta_{W_{PILOT}} \leqslant 0. \end{cases}$$

$$\delta_{SP} = \delta_{SP_R} - \delta_{SP_L}$$



K_{P_C}	0.2
K_P	2.0
KP_I	2.0
PILIM	15.0
SASALIM	15.0

Figure 5 ROLL CONTROL SYSTEM



K_{r_ϕ}	-0.15
K_{r_r}	.625
K_{P_y}	-0.5
SASRLIM	35.0

Figure 6 YAW CONTROL SYSTEM

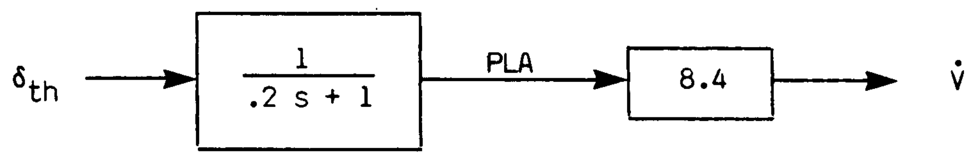
Roll Control Parameters

ω_ϕ/ω_d	1.000
ζ_ϕ/ζ_d	1.001

Time histories for a 20 degree δ_{AW} step and a 1/2 inch δ_{RP} step are shown in Figures 7 and 8.

2.2.4 Thrust

The throttle control system was the same for all of the configurations, though the response of the airplanes differed because of the different dynamics in the rest of the longitudinal axes. The pilot controlled the thrust through a throttle lever on the center console. It had a ±70% travel about its mid-trim position. It acted through a first order lag with a time constant of .2 sec to move a power lever angle (PLA) command which produced a $\dot{V}$ of ±8.4 ft/sec² at ±70%.



Time histories for 10% δ_{th} steps are shown in Appendix B for each configuration.

2.2.5 Feel System and Actuator Dynamics

The feel system parameters were chosen during the calibration flights. Known good feel system parameters from the previous TIFS/Pitch Rate program were used as the starting point and were slightly modified to provide better characteristics. Idealy the feel system should have been "transparent" to the flight test. The lack of evaluation pilot comments concerning the feel system indicates that this was achieved.

Below are the model feel system parameters that resulted from the calibration flights and were used throughout the investigation:

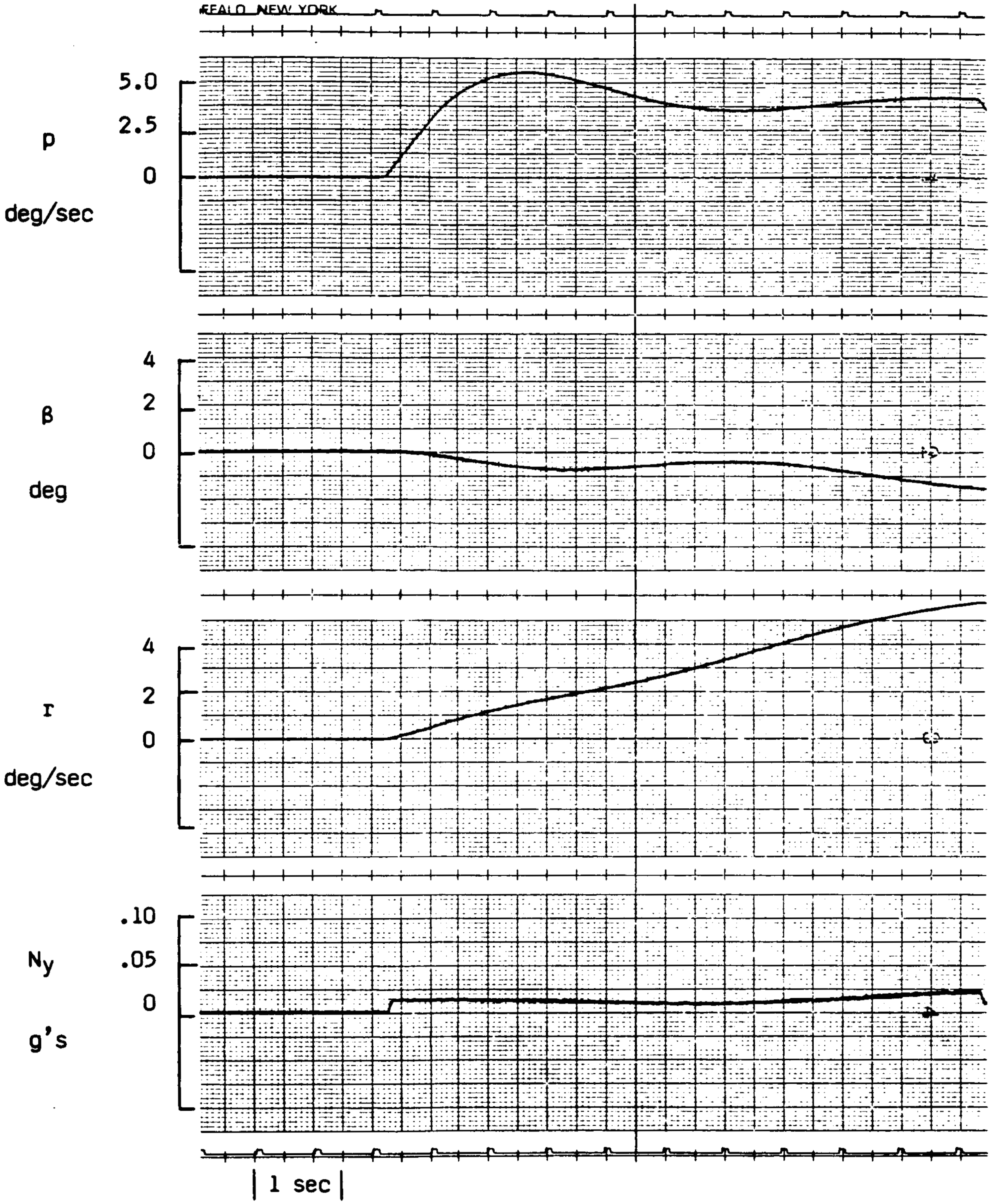


Figure 7 ROLL STEP RESPONSE, 20 DEG δ_{AW} STEP

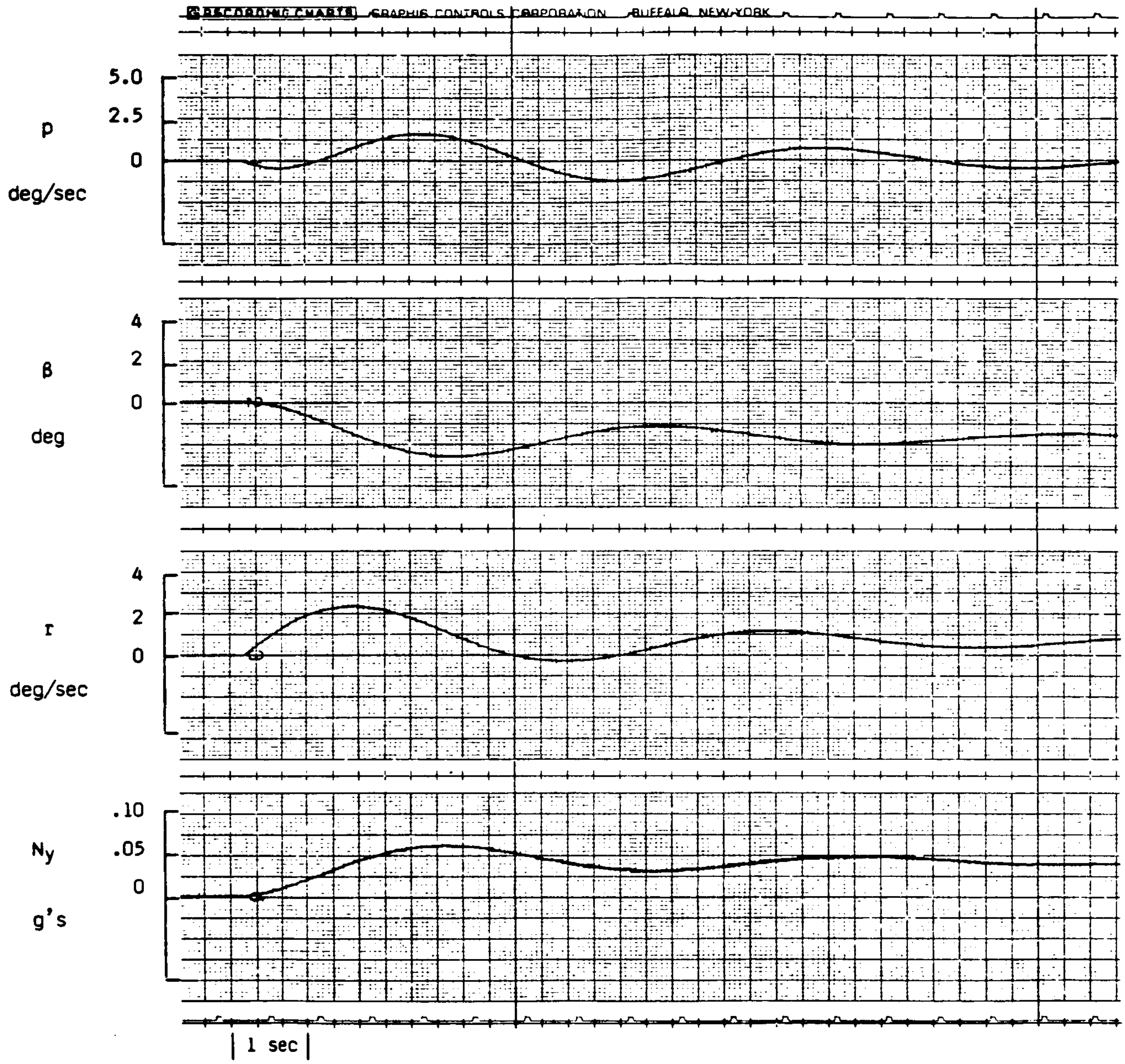


Figure 8 YAW STEP RESPONSE, ½ INCH δ_{RP} STEP

	Column Pitch	Wheel Roll	Pedals Yaw
Gradient	12 (lbs/in)	0.37 (lb/deg)	36 (lb/in)
ω_n (rad/sec)	20*	25	12
ζ	0.7	0.7	0.6
Breakout (lbs)	4.0	1.0	12

* (25 rad/sec was used in the TIFS to help reduce the effective time delay to that of the model).

The actuator dynamics used in the model for the elevator, aileron, and rudders were second order with $\omega_n = 20$ rad/sec, $\zeta = 0.7$. As previously mentioned the throttle had a first order lag with $\tau = .2$ sec.

2.2.6 Turbulence Sensitivity

External disturbances were imposed upon the evaluation in the form of a discrete vertical gust (1-cosine with a maximum amplitude of 7.6 ft/sec or $\alpha_{GUST} = 1.9$ deg with a duration of 4 sec). To make all of the configurations respond to this discrete gust in a consistant manner, a constant turbulence sensitivity vector was chosen for the α_{GUST}. The values chosen were those for the NASA TCV/737 with neutral static margin (Reference 15). The following turbulence sensitivity terms were added to the state equations:

$$\begin{bmatrix} \dot{q} \\ \dot{\theta} \\ \dot{\alpha} \\ \dot{V} \end{bmatrix} = \begin{bmatrix} 0 \\ 0 \\ -.7 \\ .4 \end{bmatrix} -\alpha_{GUST}$$

Though the turbulence sensitivities were the same for all configurations, the responses differed because of different free responses of each configuration. Any natural turbulence, if present, was not fed to the model, but did disturb the model-following system primarily in the heave axis where it was felt as a light chop and was seen in the airspeed indicator which used the TIFS airspeed. Most of the flights had light or no turbulence, so it was not a factor.

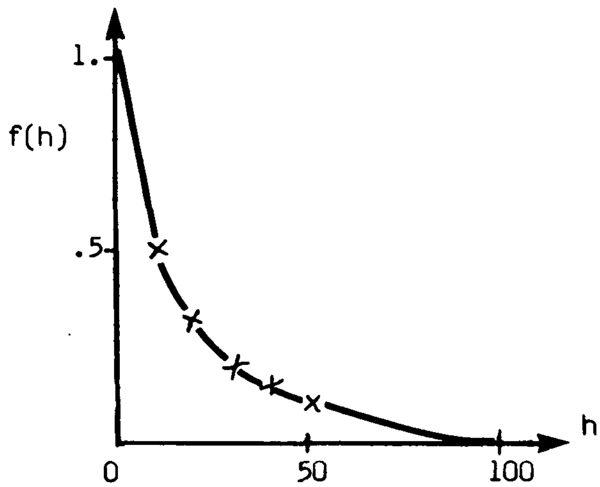
2.2.7 Ground Effect

The ground effect chosen for the model was similar to that used on the previous TIFS/Pitch Rate program, though no X-force due to ground effect was used. The following ground effect sensitivity terms were added to the state equations:

$$\begin{bmatrix} \dot{q} \\ \dot{\theta} \\ \dot{\alpha} \\ \dot{V} \end{bmatrix} = \begin{bmatrix} -.08 \\ 0 \\ -.25 \\ 0 \end{bmatrix} \cdot f(h)$$

where f(h) was the following ground effect function:

h	f(h)
≥100	0
50	.1
40	.14
30	.2
20	.32
10	.5
≤0	1



2.2.8 Bank Compensation

The steady state forces to hold a given normal acceleration varied greatly from one configuration to another. To relieve pilot pitch forces in turns and remove this factor from the evaluation which was primarily a longitudinal task, an automatic altitude-hold-in-turns system was used. For the configurations which had actual pitch rate feedback and an integrator (Configurations 17, 17 + Lead/Lag, 18-20, 21A-28A), this system was essentially a $(-g/V)\ \phi^2$ feedback to the pitch command summer. This had been used in the previous TIFS/Pitch Rate program. For the balance of the configurations, which did not have an integrator in the loop, the elevator command to hold the proper normal acceleration for a level turn as a function of bank angle had to be determined. The following derivation shows how this elevator required in turns or bank compensation was calculated.

BANK COMPENSATION

(1) $$\dot{\alpha} = Z_q q + Z_\theta \theta + Z_\alpha \alpha + Z_u u + Z_{\delta_e} \delta_e + (\cos \phi - 1) \frac{g}{V} (57.3)$$

(2) $$\dot{q} = M_q q + M_\theta \theta + M_\alpha \alpha + M_u u + M_{\delta_e} \delta_e$$

in steady turn $\dot{\alpha} = 0, \dot{q} = 0$

$$q = (57.3) \frac{g}{V} \left(n - \frac{1}{n}\right) \qquad n = (\text{load factor}) \frac{1}{\cos \phi}$$

From Equation (1)

(3) $$\alpha = \frac{-Z_q q - Z_\theta \theta - Z_u u - Z_{\delta_e} \delta_e - (\cos \phi - 1) \frac{g}{V} (57.3)}{Z_\alpha}$$

From Equation (2)

(4) $$\delta_e = \frac{-M_q q - M_\theta \theta - M_\alpha \alpha - M_u u}{M_{\delta_e}}$$

Substituting $q = \frac{g}{V} \left(n - \frac{1}{n}\right) 57.3$

(5) $$\delta_e = \frac{-M_q \left(\frac{g}{V} \left[n - \frac{1}{n}\right]\right) 57.3 - M_\theta \theta - M_\alpha \alpha - M_u u}{M_{\delta_e}}$$

Inserting Equation (3) into Equation (5) yields

$$M_{\delta_e}\delta_e = -M_q\left(\frac{g}{V}\left[n - \frac{1}{n}\right]\right) 57.3 - M_\theta\theta - M_u u - M_\alpha \frac{(-Z_q q - Z_\theta\theta - Z_u u - Z_{\delta_e}\delta_e - (\cos\phi - 1)\frac{g}{V} 57.3)}{Z_\alpha}$$

Gather δ_e terms on left side of equation

$$M_{\delta_e}\delta_e - \frac{M_\alpha}{Z_\alpha} Z_{\delta_e}\delta_e = (M_{\delta_e} - \frac{M_\alpha}{Z_\alpha} Z_{\delta_e})\,\delta_e$$

$$\delta_e = \frac{-M_q\left(\frac{g}{V}\left[n - \frac{1}{n}\right]\right)57.3 - M_\theta\theta - M_u u - \frac{M_\alpha}{Z_\alpha}\left(-Z_q\left(\frac{g}{V}\left[n - \frac{1}{n}\right]\right)57.3 - Z_\theta\theta - Z_u u - (\cos\phi - 1)\frac{g}{V} 57.3\right)}{M_{\delta_e} - \frac{M_\alpha}{Z_\alpha} Z_{\delta_e}}$$

$$\delta_e = \frac{\left[-M_q + \frac{M_\alpha}{Z_\alpha} Z_q\right]\left[\frac{g}{V}(n - \frac{1}{n})\right]57.3 - M_\theta\theta - M_u u + \frac{M_\alpha}{Z_\alpha} Z_\theta\theta + \frac{M_\alpha}{Z_\alpha} Z_u u + \frac{M_\alpha}{Z_\alpha}(\cos\phi - 1)\frac{g}{V} 57.3}{M_\delta - \frac{M_\alpha}{Z_\alpha} Z_{\delta_e}}$$

Make following assumptions: $\Delta\theta$ in turns is very small
Δu in turns is very small

so drop out $M_\theta\theta$, $M_u u$, $Z_\theta\theta$, $Z_u u$

$$\delta_e = \frac{\left[-M_q + \frac{M_\alpha}{Z_\alpha} Z_q\right]\left[\frac{g}{V}(n - \frac{1}{n})\right] 57.3 + \frac{M_\alpha}{Z_\alpha}(\cos\phi - 1)\frac{g}{V} 57.3}{M_{\delta_e} - \frac{M_\alpha}{Z_\alpha} Z_{\delta_e}}$$

$$\delta_e = \frac{-57.3\frac{g}{V} M_q (n - \frac{1}{n}) + \frac{g}{V}\frac{M_\alpha}{Z_\alpha}\left[Z_q(n - \frac{1}{n}) + 1\,(\cos\phi - 1)\right]57.3}{M_{\delta_e} - \frac{M_\alpha}{Z_\alpha} Z_{\delta_e}}$$

$$\delta_e = \frac{-57.3\,\frac{g}{V}}{Z_\alpha M_{\delta_e} - M_\alpha Z_{\delta_e}}\left[Z_\alpha M_q(n - \frac{1}{n}) - M_\alpha\left[Z_q(n - \frac{1}{n}) + 1\,(\cos\phi - 1)\right]\right]$$

$$\delta_e = \frac{-57.3\,\frac{g}{V}}{Z_\alpha M_{\delta_e} - M_\alpha Z_{\delta_e}}\left[(Z_\alpha M_q - M_\alpha Z_q)(n - \frac{1}{n}) - M_\alpha(\cos\phi - 1)\right]$$

$$n - \frac{1}{n} = \frac{1}{\cos\phi} - \cos\phi = \frac{1 - \cos^2\phi}{\cos\phi} = \frac{(1 - \cos\phi)(1 + \cos\phi)}{\cos\phi}$$

$$\delta_e = \frac{-57.3\,\frac{g}{V}}{Z_\alpha M_{\delta_e} - Z_{\delta_e} M_\alpha}\left(\frac{1 - \cos\phi}{\cos\phi}\right)\left[(Z_\alpha M_q - M_\alpha Z_q)(1 + \cos\phi) + M_\alpha\cos\phi\right]$$

$$\delta_e = -57.3\,\frac{g}{V}\left(\frac{1 - \cos\phi}{\cos\phi}\right)\left[\frac{Z_\alpha M_q - M_\alpha Z_q}{Z_\alpha M_{\delta_e} - Z_{\delta_e} M_\alpha}(1 + \cos\phi) + \frac{M_\alpha}{Z_\alpha M_{\delta_e} - Z_{\delta_e} M_\alpha}(\cos\phi)\right]$$

$$\delta_e = -K_0\left(\frac{1 - \cos\phi}{\cos\phi}\right)\left[K_1(1 + \cos\phi) + K_2(\cos\phi)\right]$$

$$K_0 = \frac{57.3\,g}{V} = 8.14$$

$$K_1 = \frac{Z_\alpha M_q - M_\alpha Z_q}{Z_\alpha M_{\delta_e} - Z_{\delta_e} M_\alpha}$$

$$K_2 = \frac{M_\alpha}{Z_\alpha M_{\delta_e} - Z_{\delta_e} M_\alpha}$$

Using the values for M_α, M_q, M_{δ_e}, Z_α, Z_q, Z_{δ_e} from the F and G matrices, the bank compensation gains K_1 and K_2 were calculated and are listed in Table 4.

Table 4

BANK COMPENSATION GAINS

Configuration	K_1	K_2
B	2.62	-1.5
1	6.79	-9.56
2	8.0	0
3	6.08	-7.97
4	6.84	2.37
5	4.28	-6.03
6	4.4	0
7	4.12	-5.44
8	4.31	.02
9	8.0	-11.88
10	8.0	0
11	6.63	-5.09
12	6.82	-8.32
13	1.98	-4.91
14	2.0	0
16	8.87	-14.32
21-28	2.62	-1.5

2.3 EVALUATION TASKS AND PROCEDURES

The evaluation task and procedures were the same as those used in the previous TIFS/Pitch Rate program, and are described below.

The evaluation pilot was given control of the aircraft on the down wind leg and performed a visual turning approach to a 1.5 to 2 mile final approach. The ILS glide slope was intercepted in the turn and held to a point 3500 ft. from the runway/glide slope intercept point. A constant speed of 132 KIAS was held thoughout the approach until landing flare.

Figure 9 details the final approach and flare geometry. A final approach "barrier" was defined as projecting up from the ground at a point 3500 ft. short of the runway and glide slope intercept point and up to the ILS glide path. The evaluation pilot was not allowed to descend below the ILS glide slope until passing the "barrier" (the position 3500 ft. short of the runway/glide slope intercept is well marked by a railroad track). Peer pressure from the safety pilots and the flight test engineer was found to be quite sufficient to prevent barrier duck under.

In addition to the altitude constraint of the barrier, lateral offsets of 200 ft. (either left or right, and obtained visually by lining up on runway markers) were used to provide secondary tasking thus preventing preoccupation with the pitch task. In order to further assure pitch task activity a (1 - cosine) angle of attack gust was fed to the model in the zone depicted in Figure 9 (between 100 and 50 feet of altitude).

The "Desired" touchdown area was defined as being 500 ft. long and 20 ft. wide (±10 ft. of centerline) starting 250 ft. past the runway/glide slope intercept. The "Adequate" touchdown area was defined as 1000 ft. long, 40 ft. wide and starting at the same point on the runway. Airspeed requirements were: "Desired" 132 ±3 KIAS, "Adequate" 132 ±5 KIAS, both at barrier passage. "Desired" sink rate at touchdown was defined as 0 to 3 fps and "Adequate" as 3 to 6 fps (these values were obtained from the data records, however, experience has shown that 0-3 fps touchdowns result in "smooth" landings, 3-6 fps touchdowns result in "solid" landings, and touchdowns in excess of 6 fps can be recognized by any crew member with a 95% confidence level).

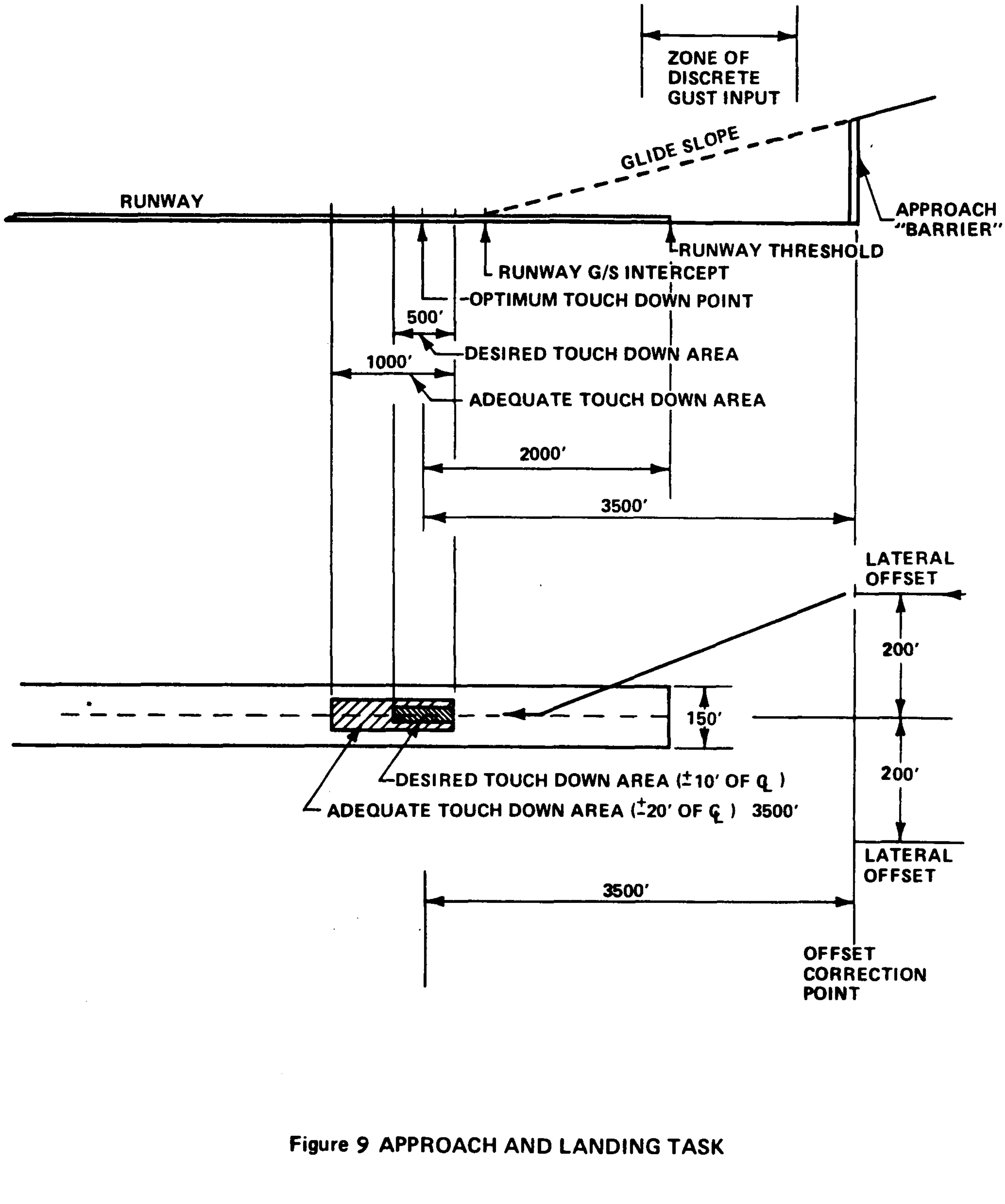


Figure 9 APPROACH AND LANDING TASK

Touchdown parameters were obtained from on-board recordings using DME and Localizer Deviation for x and y distance on the runway, and as described above for sink rate. Speeds were taken by the test engineer in the right evaluation seat and also recorded. These combined with the evaluation pilots comments and ratings provided the evaluation data.

The design goal of the above task was to achieve sufficient pilot gain in the pitch axis to provide an adequate spread in the Handling Qualities Ratings (HQR's) but not to be so difficult or easy so as to bias the HQR's one way or another.

2.3.1 Evaluation Sequence

The evaluation pilot was given control of the aircraft on the downwind leg. At this time automatic step inputs were made in order to verify the configuration. The evaluation pilot would then conduct a visual turning approach to intercept the glide slope approximately two miles from touchdown. He would visually line up for the offset and continue down the glide slope. At the "barrier" position (3500 ft. from touchdown and at approximate altitude of 200 ft.) the correction for the offset would commence and thrust and pitch would be adjusted for landing. At approximately 2000 ft. from touchdown a discrete (1 - cosine) angle of attack gust would be fed to the aircraft model to cause a standard flight path disturbance. The evaluation pilot would fly through the gust to flare and touchdown. At touchdown the safety pilots would take control of the aircraft (or at any time prior to touchdown if dictated by the situation). At this point the flight test engineer would record speeds and estimated touchdown dispersion and the evaluation pilot would begin his comments and give the HQR. This data was manually recorded by the flight test engineer as well as on voice tape. The safety pilots then executed the climbout while the TIFS technical crew set up the next configuration (if required) to repeat the process.

A normal evaluation consisted of two approaches, however, the evaluation pilot had the option of repeats if desired.

2.4 PILOT COMMENT CARD AND RATING SCALES

The evaluation pilots were briefed on the general experiment purpose and flight task details. They had a general knowledge of what the test configurations were (had seen descriptions and time histories) but no knowledge of which of those configurations would be given on each flight.

Pilot technique was necessarily different for pitch rate command flight control systems than for "conventional" systems (α-command) that require increasing average pull forces in the landing flare (i.e., monotonic). Consideration was given to informing the evaluation pilots, before hand, which type of system they had. It was decided to proceed with a "blind" experiment as it had been found successful in the previous TIFS/Pitch Rate program. The pilots adapted to technique changes rapidly and in some cases were unaware of using different techniques.

An evaluation normally consisted of two approaches and landings. The pilot could make comments at any time, however, formal use of the comment card (Figure 10), Cooper-Harper scale (Figure 11), and the PIO scale (Figure 12) was made after the second landing for the configuration. The pilot had the option of a third landing on a configuration and in that case the comments and gradings were made after the third landing.

The pilot comments and ratings were considered the primary data of the investigation, and were recorded on voice tape. In addition, the flight test engineer (in the right evaluation seat) manually recorded comment summaries, touchdown dispersion, and pilot ratings for use in the post-flight debriefing where pilot comments were elaborated in more detail.

TIFS/FLARED LANDING PROGRAM

PILOT COMMENT CARD

A. Initial Overall Impression

B. Approach

1. Initial/Final response to control inputs
2. Flight path control
3. Pitch attitude control
4. Airspeed control
5. Offset correction
6. Atmospheric disturbances
7. Special pilot techniques

C. Flare and Touchdown

1. Pitch attitude and flight path control
2. Control of touchdown parameters
3. Atmospheric disturbances
4. Special pilot techniques

D. Pilot Ratings

1. Approaches
2. Flare and touchdown
3. Overall
4. PIO rating

Figure 10 PILOT COMMENT CARD

HANDLING QUALITIES RATING SCALE

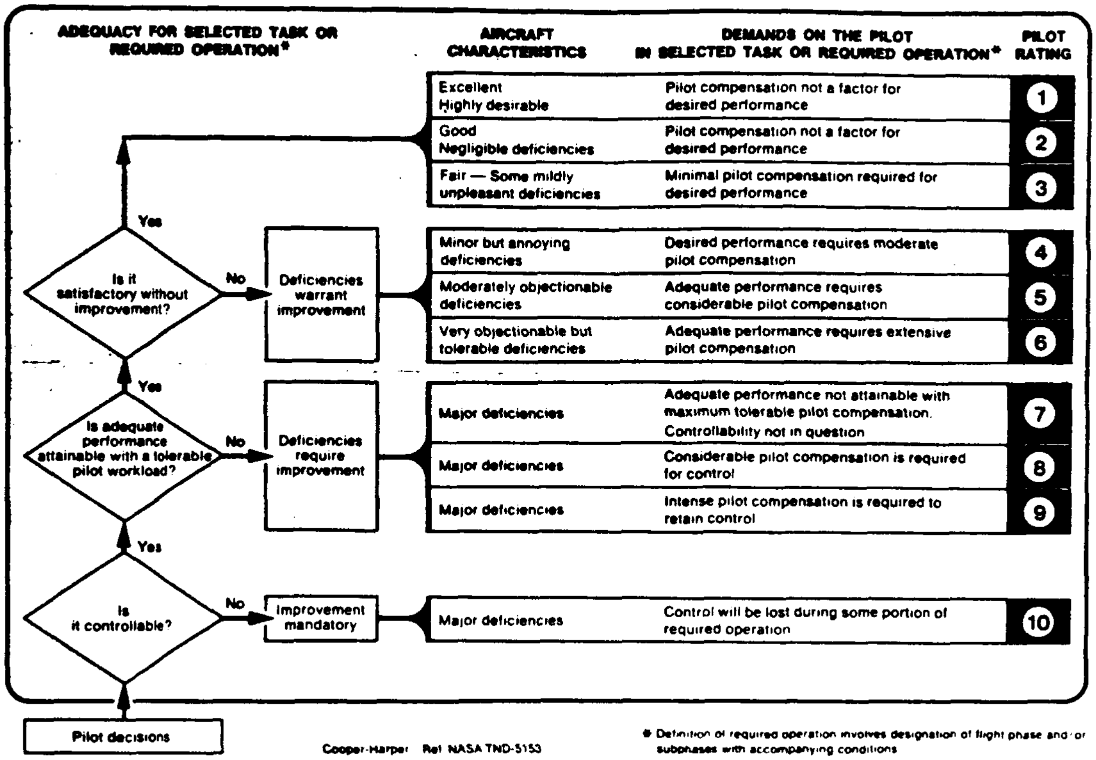


Figure 11 COOPER-HARPER HANDLING QUALILTIES RATING SCALE

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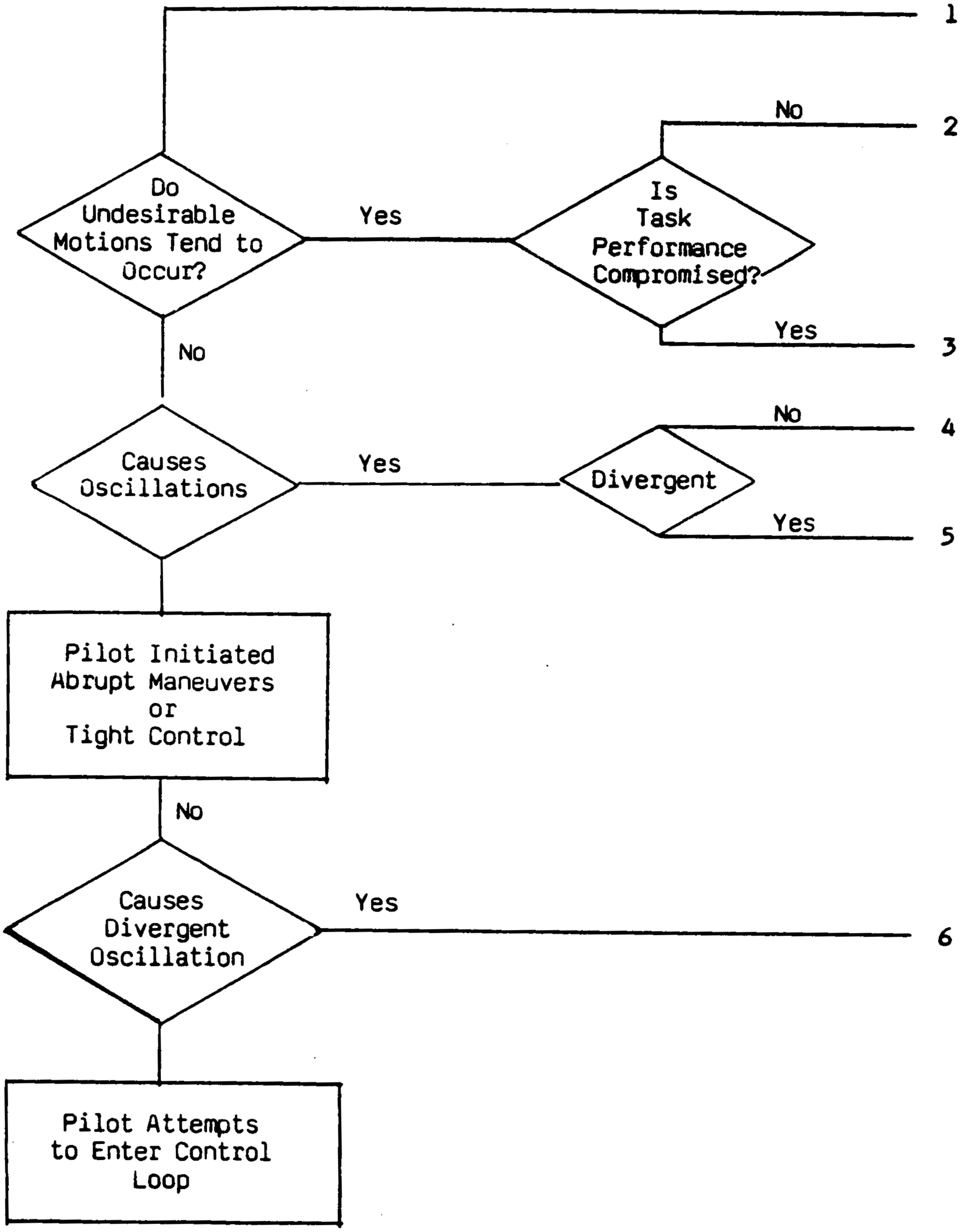

Figure 12 PIO TENDENCY CLASSIFICATION

2.5 EVALUATION PILOTS AND FLIGHT SUMMARY

This flight program was extremely fortunate to have the services of eight evaluation pilots with a wide variety of backgrounds. In addition to the NASA and Calspan pilots who were scheduled to participate in this NASA sponsored program, NASA invited other flight test organizations to provide pilots if they would pick up the costs for their flights. Boeing, Lockheed, and the German Aerospace Research Establishment (DFVLR) accepted this invitation and although their flights were conducted under a separate contract the results of their flights are presented in this report.

The evaluation pilots were:

Charles J. Berthe, Jr. -

Calspan checkout pilot (also co-investigator and safety pilot during evaluation phase) flew the calibration flights which determined the accuracy of the simulation, the efficacy of the task, fine tuned the command gains, time delays, and lateral/directional axes for the evaluation phase.

Lee H. Person, Jr. -

NASA/Langley evaluation pilot (A) - flying experience with aircraft handling qualities and IFR display programs including NASA's Terminal Configured Vehicle - 737.

John F. Ball -

Calspan evaluation pilot (B) - flying experience as an evaluation pilot in many handling qualities programs, also a safety pilot in the USAF/TIFS, NT-33, and Calspan Learjet in-flight simulators.

Roger E. Smith -

NASA/Dryden evaluation pilot (C) - flying experience as an evaluation pilot in many handling qualities programs at Calspan and NASA, also test pilot on X-29 and AFTI/F-111.

Dale M. Ranz -

Boeing evaluation pilot (D).

J. Kenneth Higgins -

Boeing evaluation pilot (E)

- Production and research test pilots for all Boeing commercial transports including evaluation of advanced flight control concepts.

Frank Hadden -

Lockheed evaluation pilot (F) - Chief Engineering Test Pilot at Lockheed - Georgia, with extensive flight test expeirence with transport category aircraft including the Jet Star C-130, C-141, C-5, and High Technology Test Bed (C-130).

Hans Meyer -

DFVLR (Germany) evaluation pilot (G) - Chief Test pilot at DFVLR with extensive flight test experience, including participation in in-flight simulation programs with the DFVLR-Hansa Jet.

Most of the evaluation approaches were conducted at Niagara Falls AFB, with approximately one-fourth of the evaluations conducted at Buffalo due to traffic and weather constraints at Niagara. There was no discernible differences in the evaluations at the two airports, but x-runway distance was not as well marked at Buffalo.

A summary of the flights versus pilot is shown in Table 5.

Table 5

FLIGHT SUMMARY

PILOT	FLIGHTS	EVALUATIONS	APPROACHES	FLT. HRS.
Calibration	3	-	27	6.1
A	7	32	67	9.1
B	6	23	53	6.8
C	4	12	21	3.8
D	1.5	7	15	1.9
E	1.5	7	17	2.0
F	3	11	22	2.9
G	3	14	29	3.9
Ferry	2	-	-	.4
Totals	31	106	251	36.9

Section 3
EXPERIMENT MECHANIZATION

3.1 EQUIPMENT

The USAF Total In-Flight Simulator (TIFS was used as the test vehicle in this experiment. TIFS is a highly modified C-131 (Convair 580) configured as a six-degree-of-freedom simulator (Figure 13). It has a separate evaluation cockpit forward and below the normal C-131 cockpit. When flown from the evaluation cockpit in the simulation or fly-by-wire mode, the pilot control commands are fed as inputs to the model computer which calculates the aircraft response to be reproduced. These responses, along with TIFS motion sensor signals, are used to generate feedforward and response error signals, which drive the six controllers on the TIFS (Figure 14). The result is a high fidelity reproduction of the motion and visual cues at the pilot position of the model aircraft. More descriptions of the TIFS can be found in Reference 16.

This experiment made use of the following major features inherent in the TIFS aircraft:

1. Independent control of all six forces and moments by use of elevator, aileron, rudder, throttle, direct lift flaps and side force surfaces.

2. Longitudinal and lateral/directional model-following systems to provide the evaluation pilot with motion and visual cues representative of the simulated aircraft.

3. Separate evaluation cockpit capable of accepting appropriate pilot controls, displays, and co-pilot assistance. (An observer but never co-pilot, was present there).

4. Evaluation cockpit instruments included standard IFR instrument displays featuring an ADI and an HSI as the primary instruments, with angle of attack and slideslip displayed on indicators to the right hand side of the HSI.

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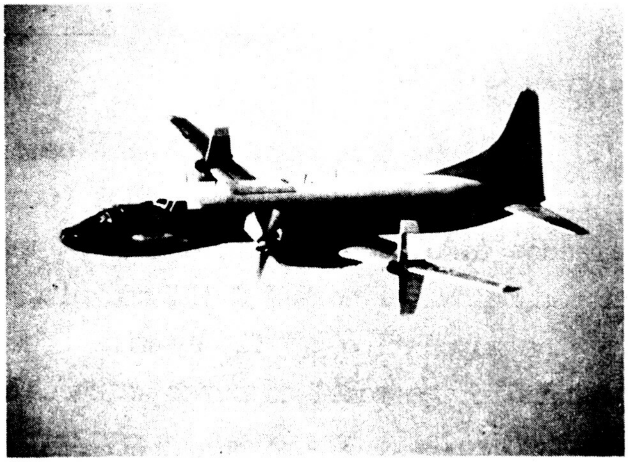

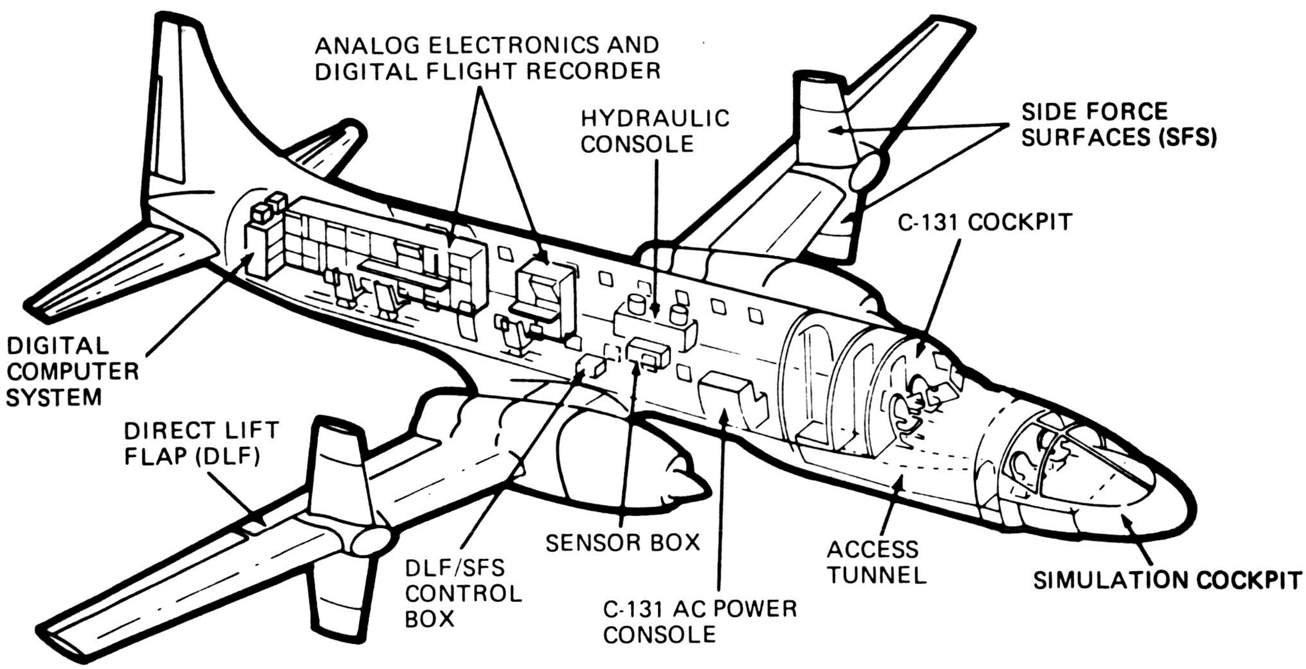

Figure 13 USAF/CALSPAN TOTAL IN-FLIGHT SIMULATOR (TIFS)

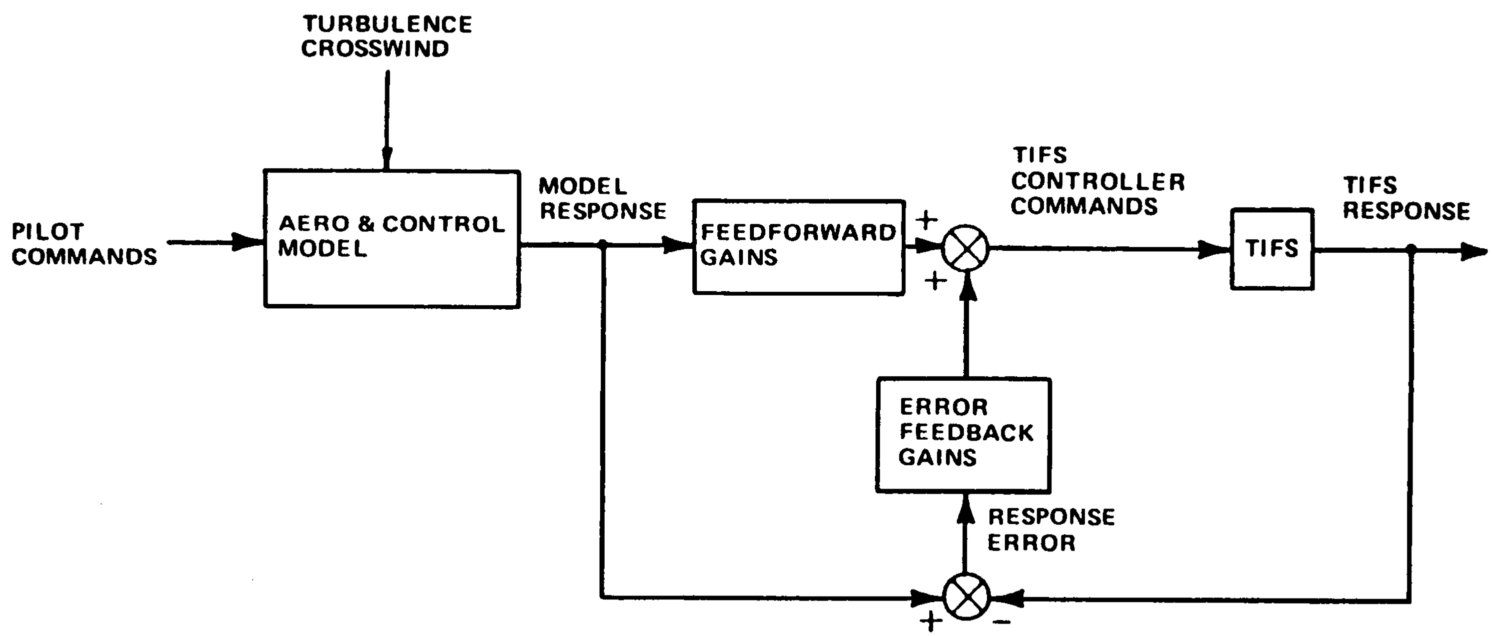

Figure 14. TIFS MODEL FOLLOWING SIMULATION

The vertical and horizontal bars on the ADI displayed command information for tracking localizer and glide slope, respectively.

5. Digital magnetic tape recording system to record control inputs and appropriate aircraft responses.

6. Two cassette tape voice recorders for recording evaluation pilot comments, and TIFS crew comments.

7. A video camera pointing out the front window and video cassette recorder to record the forward view.

8. The capability to simulate artificial or cancel actual crosswinds up to 15 kts incorporated in the model-following system.

9. Turbulence simulated by playing pre-recorded random signals into the model through filters mechanized to produce the proper power spectrum of turbulence. (Used only on two evaluations by Boeing pilot.)

10. A signal light located above the ADI and audio signal to indicate simulated or actual touchdown of main landing gear.

11. Adjustable transport time delay circuits, available to simulate time delay in the pilot's commands to the elevator and aileron controls.

12. Digital computing equipment to calculate model aerodynamics and evaluate kinematic equations.

3.2 SIMULATION GEOMETRY

The TIFS motion was configured to reproduce model motion at the evaluation pilot's eye point. As this was a generic rather than specific simulation it was decided to superimpose the TIFS c.g. and Generic Transport c.g. Additionally, the cockpits of the two aircraft were superimposed. Consequently, no transformations were required from TIFS c.g. to model c.g.

Approaches were made to touchdown and TIFS wheels and Generic Transport wheels were superimposed. This simplified geometry negated the requirement for eye position and wheel height transformations.

3.3 EVALUATION COCKPIT CONFIGURATION

The evaluation cockpit was configured as illustrated in Figure 15. The controls were standard wheel and rudders. No window masking was used. Thrust was controlled by four throttle levers tied together and total thrust was indicated on a single gage. Asymmetric thrust control was not provided.

The evaluation pilot's instrument panel is also shown in Figure 15. It was a standard configuration with raw glide slope and localizer data driving the flight director needles on the ADI.

TIFS evaluation cockpit is a dual pilot side by side arrangement. For this investigation the right seat was occupied by a NASA flight test engineer. The engineer observed all approaches and landings, assisted in conduct of the flight test card, recorded touchdown dispersion, and recorded summaries of evaluation pilot comments and handling qualities ratings (HQR) to provide timely post flight analysis.

3.4 TIME DELAY COMPENSATION

The TIFS model-following simulation mechanization has some inherent delays or lags due to computation delays, actuator dynamics, and structural dynamics. If no time delay compensation is done, the TIFS response will lag the model's response (depending on frequency content) by approximately 60 to 100 msec in pitch, 100 to 140 msec in roll and 100-140 msec in yaw. Two methods can be used to eliminate these lags. One is to use feedforward

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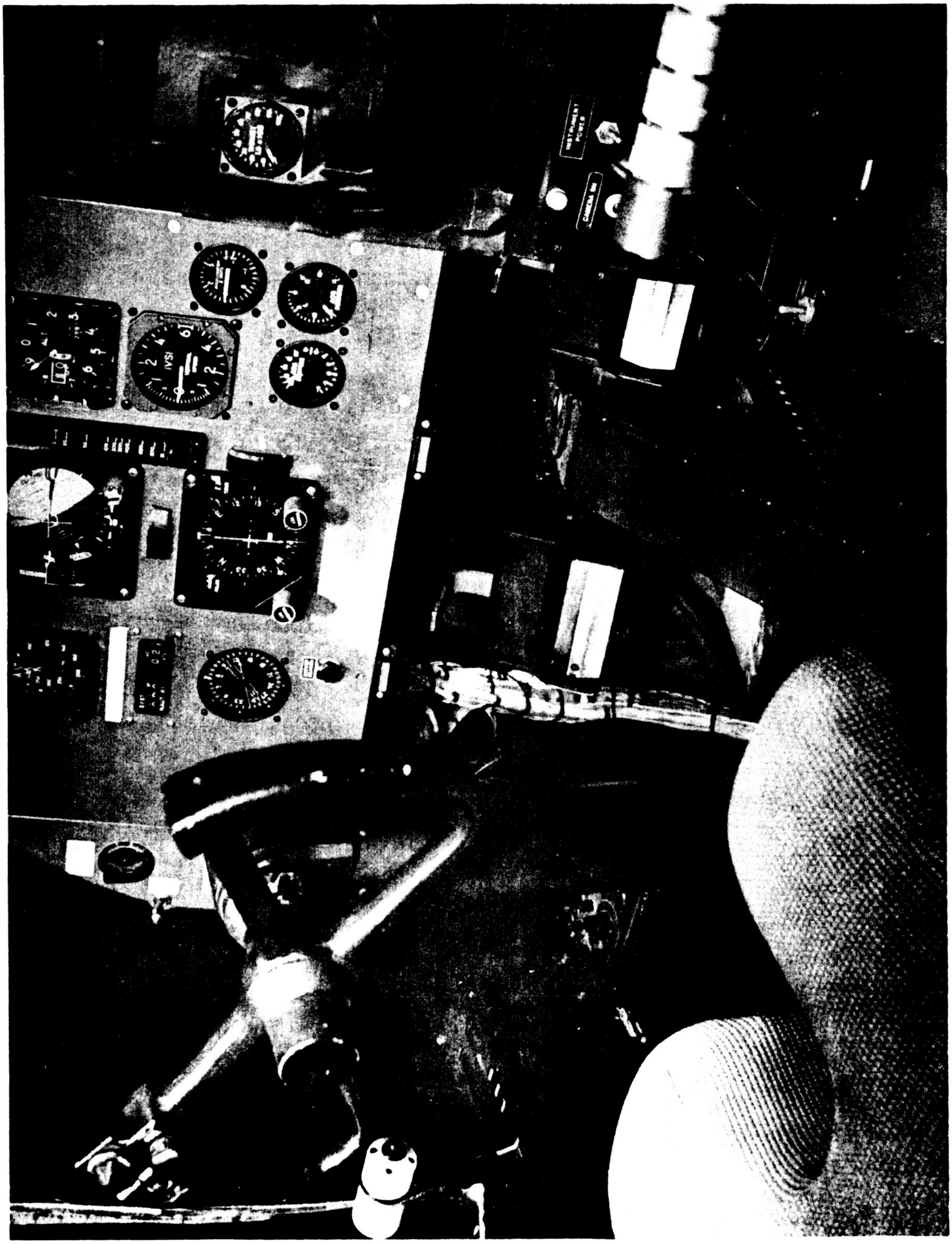


Figure 15 EVALUATION PILOT CONTROLS AND INSTRUMENT PANEL

signals from the evaluation pilot's inputs or model actuator inputs directly to the TIFS command actuators. This will start to command TIFS surface motions before model repsonses have been generated. If these feedforward "jabber" signals are properly tuned, one can eliminate the lag without significantly changing the response shape. The second method that can be used is to eliminate delays or lags in the model itself so that it responds faster than the actual aircraft being modeled. When the TIFS' lags are added to the speeded up model the final response occurs at the proper time. This latter method was used in the present simulation because of its ease of implementation.

In the pitch axis a 60 msec pure time delay was used in the analytical model but not in the model mechanized on the TIFS computer. In addition no pitch actuators were used in the TIFS model mechanization. The result was a TIFS pitch rate response which matched the time delay characteristics of the previously defined analytical model.

In the heave axis the TIFS direct lift flaps have a higher bandwidth than the elevators (approximately 44 rad/sec versus 20 rad/sec). In order to slow down the direct lift flaps so that they would be "in-phase" with the elevators, a first order lag was inserted in the feedforward commands to the direct lift flaps. This lag had a time constant of .038 sec and slowed down the flaps to effectively be "in-phase" with the elevator.

No time delay compensation was done in the lateral/directional axes. The lateral/directional responses were considered to have Level 1 flying qualities in the last TIFS/ Pitch Rate program without compensation. This was again verified in the checkout phase where slight modifications were made in the roll axis feel system to make the forces consistent with a transport aircraft feel.

3.5 MODEL FOLLOWING VERIFICATION

Step input responses were taken in flight for all of the configurations for comparison to the analytical model responses to validate the simulation. Pitch as well as roll and yaw step responses are shown in Appendix C. It can be seen that fairly good overlays were obtained. The only characteristic of note is the 3.4 Hz oscillation in N_{z_p} that occurs with the very abrupt configurations (especially 2 and 4). This is from a lightly damped structural mode. They appear large, but are being superimposed on a fairly small steady state incremental acceleration (~.1g). The magnitude of the structural oscillation remained the same with larger inputs. This type of structural oscillation also appears when flying in turbulence. It was not considered to be a factor in the evaluations as the pilots never noted the oscillatory characteristics.

Typical model following responses during approaches are shown in Figures 16 and 17. In these figures the TIFS response is the solid line and the model response is the dashed line.

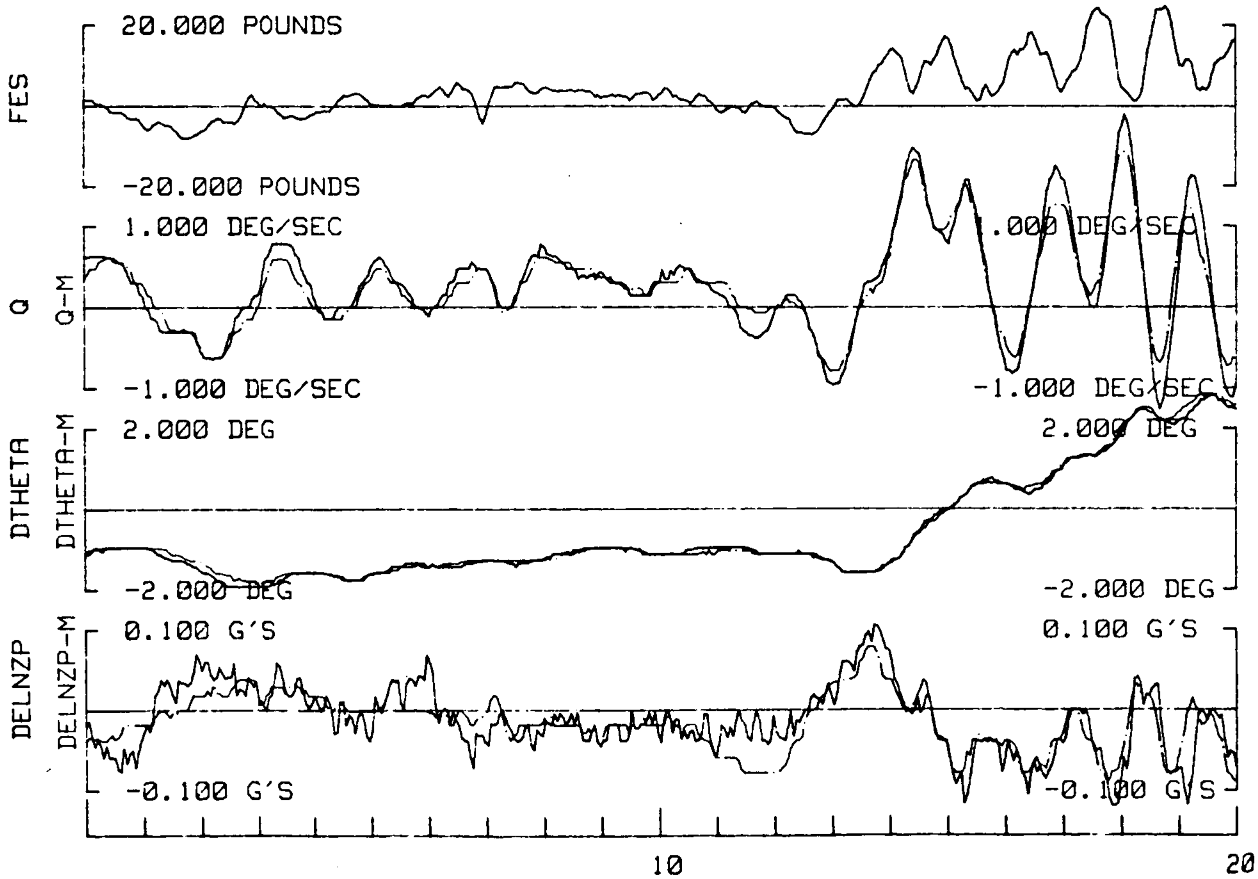


Figure 16 MODEL FOLLOWING - FLIGHT 893, APPROACH 1, CONFIGURATION B

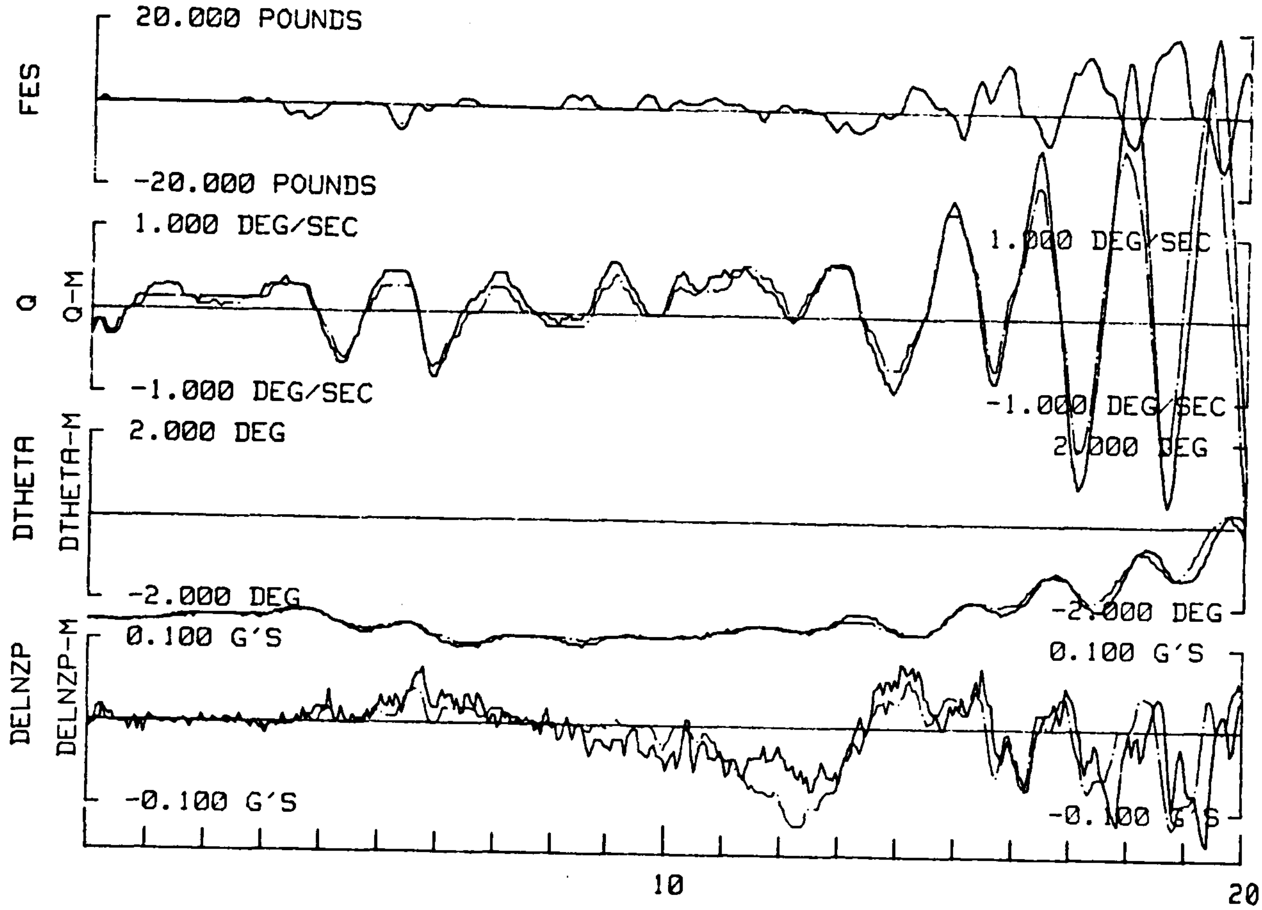


Figure 17 MODEL FOLLOWING - FLIGHT 893, APPROACH 10, CONFIGURATION 28

3.6 DATA RECORDING

Quantitative data was recorded on board the TIFS on a 58 channel digital recorder. Table 6 is the recording list of this data. Signals include TIFS responses (unsubscripted), model responses (M-subscript), pilot inputs, control surface positions and performance parameters. Signals which are increments from the engage value are indicated by a Δ. All angular units are in degrees, velocity in ft/sec (except as noted), accelerations in g's.

Two voice recorders were utilized to obtain evaluation pilot comments. One was generally used just for the evaluation pilot (plus anyone on hot microphone) and the other recorded all crew comments and radio transmissions.

A video camera was installed in the evaluation cockpit and recorded the forward field of view.

Table 6

TIFS RECORDING LIST

CHANNEL	VARIABLE	
1	q_M	- pitch rate, MODEL
2	q	- pitch rate, TIFS
3	p_M	- roll rate, MODEL
4	p	- roll rate, TIFS
5	r_M	- yaw rate, MODEL
6	r	- yaw rate, TIFS
7	$\Delta\alpha_M$	- incremental angle of attack, MODEL
8	$\Delta\alpha$	- incremental angle of attack, TIFS
9	β_M	- sideslip, MODEL
10	β	- sideslip, TIFS
11	$\Delta\theta_M$	- incremental pitch attitude, MODEL
12	$\Delta\theta$	- incremental pitch attitude, TIFS
13	ϕ_M	- roll angle, MODEL
14	ϕ	- roll angle, TIFS
15	$\Delta N_{z_{p_M}}$	- incremental normal acceleration, pilot, MODEL
16	ΔN_{z_p}	- incremental normal acceleration, pilot, TIFS
17	ΔN_{z_M}	- incremental normal acceleration, CG, MODEL
18	ΔN_z	- incremental normal acceleration, CG, TIFS
19	N_{y_M}	- lateral acceleration, MODEL
20	N_y	- lateral acceleration, TIFS
21	ΔV_M	- incremental true airspeed, MODEL
22	ΔV	- incremental true airspeed, TIFS
23	$\dot{h}$	- altitude rate, TIFS
24	ERR-δ_z	- error in δ_z (internal model following signal)

Table 6 (Cont'd)
TIFS RECORDING LIST

CHANNEL	VARIABLE	
25	γ	- flight path angle, TIFS
26	V_{IAS}	- indicated airspeed, knots, TIFS
27	V_M	- total true airspeed, MODEL
28	α_M	- total angle of attack, MODEL
29	α_g	- angle of attack, gust component added to Model
30	$\dot{\alpha}_M$	- angle of attack rate, TIFS
31	h_{WH}+ pulse$_{TD}$	- altitude of wheels plus pulse at touchdown
32	h (press)	- pressure altitude
33	Loc. Dev.	- localizer deviation
34	GS Dev.	- glide slope deviation
35	F_{EC}	- pitch column force
36	F_{AW}	- roll wheel force
37	F_{RP}	- rudder pedal force
38	δ_{EC}	- pitch column deflection, inch
39	δ_{AW}	- roll wheel deflection, deg
40	δ_{RP}	- rudder pedal deflection, inch
41	PLA	- power lever angle, %
42	δ_{e_M}	- elevator position, MODEL
43	δ_{a_M}	- aileron position, MODEL
44	δ_{r_M}	- rudder position, MODEL
45	$\dot{h}_M$	- altitude rate, MODEL
46	$\Delta\alpha_c$	- incremental angle of attack, complimentary filtered, TIFS
47	$\delta_{z_{FF}}$	- feedforward command to δ_z, TIFS
48	$\dot{\alpha}_M$	- angle of attack rate, MODEL

Table 6 (Cont'd)
TIFS RECORDING LIST

CHANNEL	VARIABLE		
49	γ_P	-	flight path angle, PILOT
50	CONF	-	digital configuration number
51	y	-	lateral distance on runway (+ left of center)
52	DME	-	distance from tacan, nautical miles
53	δ_e	-	elevator, TIFS
54	δ_a	-	aileron, TIFS
55	δ_r	-	rudder, TIFS
56	δ_z	-	direct lift flap, TIFS
57	δ_y	-	sideforce surface, TIFS
58	δ_x	-	throttle, TIFS

Section 4
DATA

4.1 INTRODUCTION

This section presents the data obtained from the evaluation flights. Included is a chronology of the evaluations, pilot ratings and performance. Pilot comments and approach time histories are presented in the appendices. Analysis dealing with the commanded response and time delay/ sensitivity aspects of the program are presented in Sections 5 and 6.

4.2 EVALUATION CHRONOLOGY

After a set of three checkout flights which took place during late November and early December of 1985, the evaluation flights commenced. A typical evaluation flight day started with a takeoff from Buffalo and proceeding to Niagara where evaluation approaches took place. After a configuration was set up by the test engineer, the system was engaged, responses to an automatic step input were recorded for verification, and then control was given to the evaluation pilot. The final full stop landing was made at Niagara if another flight was going to be flown. A post flight debriefing was held during the refueling operation. After the final evaluation at Niagara, the TIFS proceeded to Buffalo for the final landing of the day. This generally was not an evaluation. On two days a separate non-evaluation ferry flight had to be flown from Niagara to Buffalo.

During December of 1985 all of the evaluation flights for Pilot B and most of the evaluation flights for Pilots A and C were flown. In addition the pilots from Boeing (D and E) and the pilot from DFVLR (G) had their evaluation flights. In January of 1986 the evaluation flights for pilot A and C were completed as well as those for the Lockheed pilot (F). Most of these January flights were flown at Buffalo because of poor weather and visibility at Niagara.

A summary of the evaluation flight data in chronological order is presented in Table 7. Included in this listing are:

Flight number
date
level of turbulence
a note if approaches were at Buffalo

Pilot:	A	Person	-	NASA/Langley
	B	Ball	-	Calspan
	C	Smith	-	NASA/Dryden
	D	Ranz	-	Boeing
	E	Higgins	-	Boeing
	F	Hadden	-	Lockheed
	G	Meyer	-	DFVLR

Configuration: CONF

Pilot Ratings: separately listed for approach portion (APP), flare and touchdown (TD), overall (OA).

Pilot Induced Oscillation Rating: PIOR

Approach number giving sequence order on flight. Approaches not listed were aborted due to traffic.

Touchdown Performance:

$-\dot{h}$ - sink rate, ft/sec

V - airspeed, knots - IAS

X - longitudinal runway position (obtained from TACAN-DME) (+) long, (-) short, ft (Resolution was 60 ft) (On a few flights, indicated by a "--", the DME was not turned on, and approaches to Buffalo did not have this measurement. On these approaches Long or Short indicates the touchdown was out of the desired zone and was called out by the flight test engineer).

Y - lateral runway position, (+) left, (-) right, ft.

Table 7
TIFS/FLARED LANDING APPROACH
EVALUATION FLIGHT DATA SUMMARY

FLT - DATE TURBULENCE	PILOT	CONF	PILOT RATING			PIOR	APP #	TD PERFORMANCE			
			APP	TD	OA			$-\dot{h}$	V	X	y
891 - 12/4/85 None	A	B	3	3	3	1	1 2 3 4	No T.D. - Heavy No T.D. - Heavy No T.D. - Heavy No T.D. - Heavy			
		1	2	$2\frac{1}{2}$	$2\frac{1}{2}$	1	5 6	4 4	118 120	0 120	-2 -2
		2	2	3	3	1	7 8	3 2	126 116	480 0	0 6
892 - 12/5/85 None	A	24	3	4	4	1	2 3	4 2	118 113	-60 0	7 3
		21	3	4	4	1	4 5	2 3	118 118	0 0	1 6
		27	3	5	5	4	6 7	1 1	116 118	0 240	12 12
		28	2	3	3	2	8 9	3 3	120 118	0 0	15 11
		26	2	$2\frac{1}{2}$	$2\frac{1}{2}$	1	10 11	1 2	116 113	0 0	14 6
		25	3	3	3	1	12 13	2 2	113 116	0 0	5 7

Table 7 (Cont'd)
TIFS/FLARED LANDING APPROACH
EVALUATION FLIGHT DATA SUMMARY

FLT - DATE TURBULENCE	PILOT	CONF	PILOT RATING			PIOR	APP #	TD PERFORMANCE			
			APP	TD	OA			$-\dot{h}$	V	X	y
893 - 12/5/85 None	B	B	2	3	3	1	1	No T.D. - Heavy			
							2	No T.D. - Heavy			
							3	3	116	0	4
							4	1	116	240	-1
							5	1	113	0	-2
		25	3	5	5	3	6	2	116	0	-7
							7	2	113	0	1
		22	7	9	9	6	8	No T.D. - In PIO			
							9	No T.D. - In PIO			
		28	7	10	10	6	10	No T.D. - In PIO			
							11	No T.D. - In PIO			
894 - 12/5/85 None	A	17	2	2	2	1	1	4	124	0	-8
							2	1	116	0	-2
		18	3	2	3	1	3	3	118	120	7
							4	1	116	0	3
		19	$2\frac{1}{2}$	$2\frac{1}{2}$	$2\frac{1}{2}$	1	5	3	114	0	4
							6	1	118	0	7
		20	2	2	2	1	7	2	115	0	20
							8	1	118	0	5
895 - 12/7/85 Light	A	3	3	3	3	1	1	1	116	0	-1
							2	1	113	0	1
		23	6	3	4	1	3	1	120	-60	-1
							4	1	120	0	-2
		22	5	7	7	4	5	1	122	0	6
							6	No T.D.-Inten. PIO			
							7	No T.D.-Inten. PIO			
		4	4	2	4	1	9	1	120	0	13
							10	1	119	0	3
		6	2	2	2	1	11	1	115	0	12
							12	1	116	60	5

Table 7 (Cont'd)
TIFS/FLARED LANDING APPROACH
EVALUATION FLIGHT DATA SUMMARY

FLT - DATE TURBULENCE	PILOT	CONF	PILOT RATING			PIOR	APP *	TD PERFORMANCE			
			APP	TD	OA			$-\dot{h}$	V	X	y
896 - 12/7/85 Light	B	3	6	6	6	1	1 2	1 1	111 114	0 60	8 4
		7	4	5	5	1	3 4 5	2 1 1	110 110 111	1020 0 0	-3 4 0
		17+ Lead/ Lag	3	3	3	1	6 7	1 1	120 122	0 60	3 -2
		24	3	3	3	1	8 9	1 1	120 118	0 60	3 6
897 - 12/7/85 Light	B	21	3	6	6	4	1 2	1 1	116 119	-- --	0 -3
		27	5	8	8	5	3 4	1 No T.D. - In PIO	125	Long	-2
		25A	4	5	5	1	5 6	1 1	121 126	Long --	4 -6
		22A	5	9	9	5	7 8	3 5	118 125	-- --	-8 -8
898 -12/10/85 None	B	23	2	3	3	1	1 2 3	1 1 1	114 118 118	1020* 480 240	-5 -5 -2
		28A	7	10	10	6	4 5	No T.D. - In PIO No T.D. - In PIO			
		26	3	4	4	1	6 7	1 1	120 120	0 480	-7 4
		8	3	2	3	1	8 9	1 1	122 120	0 120	5 4
		17	3	3	3	1	10 11	1 1	119 116	0 120	6 2

* Do not count in performance - pilot not set up

Table 7 (Cont'd)
TIFS/FLARED LANDING APPROACH
EVALUATION FLIGHT DATA SUMMARY

FLT - DATE TURBULENCE	PILOT	CONF	PILOT RATING			PIOR	APP #	TD PERFORMANCE			
			APP	TD	OA			$-\dot{h}$	V	X	y
900 -12/12/85 None	B	20	5	4	5	1	1 2	2 1	120 123	0 0	-4 8
		1	3	3	3	1	3 4	1 1	118 117	0 120	5 -4
		2	3	3	3	1	5 6	1 1	113 116	120 0	3 -3
		4	2	2	2	1	7 8	1 1	116 118	0 120	-2 2
		5	5	6	6	3	9 10	1 1	117 116	60 480	0 -2
901 -12/12/85 None (Boeing Flt.)	D	B	2	2	2	1	1 2	1 2	120 121	0 0	4 2
		25	3	4	4	1	4 5	3 3	124 123	240 0	2 0
		28	5	8	8	3	6 7	1 2	125 122	0 0	3 4
902 -12/12/85 Light (Boeing Flt.)	E	B	3	3	3	1	1 2 3	1 1 1	126 123 123	-- -- --	-1 -4 2
		24	2	3	3	1	4 5	1 2	120 123	Long --	-1 -5
		27	4	8	8	5	6 7	No T.D. - In PIO 1	 124	 --	 4
		B	3	2	3	1	8 9	1 1	118 120	-- 0	-8 3

Table 7 (Cont'd)
TIFS/FLARED LANDING APPROACH
EVALUATION FLIGHT DATA SUMMARY

FLT - DATE TURBULENCE	PILOT	CONF	PILOT RATING			PIOR	APP #	TD PERFORMANCE			
			APP	TD	OA			$-\dot{h}$	V	X	y
904 -12/13/85 None (Boeing Flt.)	E	17+ Lead/ Lag	4	3	4	1	3 4	No T.D. - Heavy 1	 125	 0	 14
		17	1	2	2	1	5 6	2 1	118 124	120 1020	15 11
		20	3	2	3	1	7 8	1 1	120 120	240 0	8 12
	D	20	4	3	4	1	9 10	1 1	114 113	480 0	13 10
with moderate taped turbulence		17	4	4	4	1	11 12	1 1	116 123	0 1020	5 7
		17+ Lead/ Lag	2	3	3	1	13 14	1 1	128 118	0 240	2 8
		16	2	2	2	1	15 16	1 1	125 125	480 60	5 6
905 -12/13/85 Light	B	6	6	3	6	1	2 4	1 1	116 123	-- 240	4 -4
906 -12/17/85 Light	C	B	3	3	3	1	1 2	1 2	119 125	0 0	0 -3
		21	3	2	3	1	3 4	1 1	123 120	0 0	-4 -2
		27	3	6	6	4	5 6	1 1	115 120	0 0	-4 1
		24	5	5	5	1	7	2	116	0	0

Table 7 (Cont'd)
TIFS/FLARED LANDING APPROACH
EVALUATION FLIGHT DATA SUMMARY

FLT - DATE TURBULENCE	PILOT	CONF	PILOT RATING			PIOR	APP *	TD PERFORMANCE			
			APP	TD	OA			$-\dot{h}$	V	X	y
907 -12/17/85	G	1	3	3	3	1	1	1	119	--	4
Light							2	2	119	Long	6
(DFVLR Flt.)							3	2	129	--	-2
All app's at		2	3	6	6	1	4	1	120	--	5
Buffalo							5	1	124	Long	0
		13	3	9	9	5	6	1	125	--	0
							7	No T.D. - In PIO			
		14	3	5	5	1	8	1	125	Long	1
							9	1	120	Long	4
908 -12/18/85 Light to Mod. 5Kt of X-wind not canceled. All app's at Buffalo	C	17	2	2	2	1	1	3	126	--	-6
							2	3	120	--	-11
909 -12/20/85	G	11	2	2	2	2	1	No T.D. - Heavy			
None							2	No T.D. - Heavy			
(DFVLR Flt.)		12	4	6	6	3	3	1	115	0	2
							4	1	123	120	9
		3	2	2	2	1	5	1	125	120	8
							6	1	117	60	6
		4	5	5	5	1	7	2	127	0	8
							8	2	125	0	6
		5	3	3	3	1	9	1	120	1020	9
							10	1	116	480	4
		6	3	3	3	1	11	1	116	480	-1
							12	1	125	240	7
		7	2	2	2	1	13	1	123	480	11
							14	1	123	0	4

Table 7 (Cont'd)
TIFS/FLARED LANDING APPROACH
EVALUATION FLIGHT DATA SUMMARY

FLT - DATE TURBULENCE	PILOT	CONF	PILOT RATING			PIOR	APP #	TD PERFORMANCE			
			APP	TD	OA			$-\dot{h}$	V	X	y
910 -12/20/85 None (DFVLR Flt.)	G	8	2	2	2	1	1 2	1 1	125 119	Long --	4 9
		9	4	5	5	1	3 4	1 1	120 116	Long --	19 9
		10	2	3	3	1	5 6	1 1	120 119	-- --	9 8
911 - 1/7/86 Light	C	26	3	3	3	1	1 2	2 1	121 113	0 0	-5 -5
		23	5	5	5	1	3	1	112	0	0
		28	3	7	7	4	4 5	1 1	118 116	0 0	-6 -7
912 - 1/7/86 Light to Mod. (Lockheed Flt) All app's at Buffalo	F	B	2	2	2	1	1 2 3	1 3 3	115 123 121	-- -- --	4 -4 -4
		24	3½	3½	3½	1	4 5	4 6	125 115	-- --	-3 -4
		21	5	5	5	3	6 7	2 2	126 121	 Long	-7 -7
Incomplete	Eval. →	27	3	3	3	1	8	3	120	--	-2
913 - 1/8/86 None (Lockheed Flt) All app's at Buffalo	F	27	2	4½	4½	4	1 2	2 3	120 125	-- Long	-4 -1
		25	5	7	7	4	3 4 5	4 2 2	125 127 116	-- Short Short	-7 -3 -5
		22	3	6	6	4	6	2	124	Long	-4

Table 7 (Cont'd)
TIFS/FLARED LANDING APPROACH
EVALUATION FLIGHT DATA SUMMARY

FLT - DATE TURBULENCE	PILOT	CONF	PILOT RATING			PIOR	APP ●	TD PERFORMANCE			
			APP	TD	OA			$-\dot{h}$	V	X	y
914 - 1/8/86 Light All app's at Buffalo	A	17+ Lead/ Lag	4	3	4	1	1 2	1 2	116 118	-- --	-4 -2
		25A	4	$3\frac{1}{2}$	4	1	3 4	1 1	114 121	-- --	-1 -4
		22A	$3\frac{1}{2}$	$4\frac{1}{2}$	$4\frac{1}{2}$	3	5 6	1 4	111 116	-- --	-6 -7
		28A	2	5	5	4	7	1	119	--	-4
915 - 1/8/86 Light to Mod. All app's at Buffalo	C	22	4	5	5	2	1 2	1 1	123 120	-- --	3 -4
		25	6	8	8	3	3 4	3 3	119 119	-- --	0 4
		4	4	4	4	2	5 6	2 2	119 125	-- --	-7 0
		8	2	2	2	1	7	3	123	--	-7
916 - 1/9/86 Light All app's at Buffalo	A	25	2	4	4	1	1 2	3 3	118 121	Short --	-5 -10
		28	3	8	8	6	3 4	No T.D. - In PIO 1	 119	 --	 -6
		5+ 200 ms.	3	9	9	6	5 6	1 No T.D. - In PIO	114	--	-7
		7	4	5	5	1	7 8	No T.D.- 1	 111	Float --	Long -13

Table 7 (Cont'd)
TIFS/FLARED LANDING APPROACH
EVALUATION FLIGHT DATA SUMMARY

FLT - DATE TURBULENCE	PILOT	CONF	PILOT RATING			PIOR	APP #	TD PERFORMANCE			
			APP	TD	OA			$-\dot{h}$	V	X	y
917 - 1/9/86 Moderate (Lockheed Flt) All app's at Buffalo	F	28	3	5	5	3	1 2	5 2	124 120	-- --	-8 0
		1	3	3	3	1	3 4	2 1	118 125	-- --	-4 -7
		13	3	$5\frac{1}{2}$	$5\frac{1}{2}$	1	5 6	4 3	130 123	-- Long	-4 -6
		11	5	8	8	4	7 8	No T.D. - In PIO 3	 120	 Long	 -1
918 - 1/10/86 Light	A	9	3	3	3	1	1 2	No T.D. - Heavy 1	 113	 0	 -2
		10	$2\frac{1}{2}$	$2\frac{1}{2}$	$2\frac{1}{2}$	1	3 4	1 3	110 116	-180 0	-1 1
		14	3	4	4	3	5 6	1 1	125 128	0 0	-2 -2
		11	5	7	7	4	7 8	1 1	117 121	0 0	-6 0
		12	6	9	9	5	9 10	No T.D. - In PIO No T.D. - In PIO			
At Buffalo		5	4	6	6	3	11 12	1 3	121 121	-- Long	-7 -6

A cross reference between configuration, flight number and pilot is provided in Table 8.

Table 8

CONFIGURATION - FLIGHT NUMBER/PILOT CROSS REFERENCE

CONFIGURATION	FLIGHT/PILOT
B	891/A, 893/B, 906/C, 901/D, 902/E, 912/F
1	891/A, 900/B, 917/F, 907/G
2	891/A, 900/B, 907/G
3	895/A, 896/B, 909/G
4	895/A, 900/B, 915/C, 909/G
5	918/A, 900/B, 909/G, (916/A - with +200 ms delay)
6	895/A, 905/B, 909/G
7	916/A, 896/B, 909/G
8	898/B, 915/C, 910/G
9	918/A, 910/G
10	918/A, 910/G
11	918/A, 917/F, 909/G
12	918/A, 909/G
13	917/F, 907/G
14	918/A, 907/G
16	904/D
17	894/A, 898/B, 908/C, 904/D, 904/E
17 + Lead/Lag	914/A, 896/B, 904/D, 904/E
18	894/A
19	894/A
20	894/A, 900/B, 904/D, 904/E
21	892/A, 897/B, 906/C, 912/F
22	895/A, 893/B, 915/C, 913/F
23	895/A, 898/B, 911/C
24	892/A, 896/B, 906/C, 902/E, 912/F
25	892/A, 916/A, 893/B, 915/C, 901/D, 913/F
26	892/A, 898/B, 911/C
27	892/A, 897/B, 906/C, 902/E, 912/F, 913/F
28	892/A, 916/A, 893/B, 911/C, 901/D, 917/F

Table 8 (Cont'd)

CONFIGURATION - FLIGHT NUMBER/PILOT CROSS REFERENCE

CONFIGURATION	FLIGHT/PILOT
22A	914/A, 897/B
25A	914/A, 897/B
28A	914/A, 898/B

4.3 PILOT RATINGS

A summary tabulation of the pilot ratings versus configuration and evaluation pilot are shown in Tables 9, 10, and 11 for Approach, Flare and Touchdown, and Overall ratings, respectively. PIO ratings are tabulated on Table 12.

Table 9

PILOT RATINGS, APPROACH

	PILOT						
CONFIGURATION	A	B	C	D	E	F	G
B	3	2	3	2	3,3	2	
1	2	3				3	3
2	2	3					3
3	3	6					2
4	4	2	4				5
5	4,3*	5					3
6	2	6					3
7	4	4					2
8		3	2				2
9	3						4
10	$2\frac{1}{2}$						2
11	5					5	2
12	6						4
13						3	3
14	3						3
16				2			
17	2	3	2	4	1		
17 + L/L	4	3		2	4		
18	3						
19	$2\frac{1}{2}$						
20	2	5		4	3		
21	3	3	3			5	
22	5	7	4			3	
23	6	2	5				
24	3	3	5		2	$3\frac{1}{2}$	
25	3,2	3	6	3		5	
26	2	3	3				
27	3	5	3		4	2	
28	2,3	7	3	5		3	
22A	$3\frac{1}{2}$	5					
25A	4	4					
28A	2	7					

* had +200 ms delay inserted inadvertantly

Table 10

PILOT RATINGS, FLARE AND TOUCHDOWN

	PILOT						
CONFIGURATION	A	B	C	D	E	F	G
B	3	3	3	2	3,2	2	
1	2½	3				3	3
2	3	3					6
3	3	6					2
4	2	2	4				5
5	6,9*	6					3
6	2	3					3
7	5	5					2
8		2	2				2
9	3						5
10	2½						3
11	7					8	2
12	9						6
13						5½	9
14	4						5
16				2			
17	2	3	2	4	2		
17 + L/L	3	3		3	3		
18	2						
19	2½						
20	2	4		3	2		
21	4	6	2			5	
22	7	9	5			6	
23	3	3	5				
24	4	3	5		3	3½	
25	3,4	5	8	4		7	
26	2½	4	3				
27	5	8	6		8	4½	
28	3,8	10	7	8		5	
22A	4½	9					
25A	3½	5					
28A	5	10					

* had +200 ms delay inserted inadvertantly

Table 11

PILOT RATINGS, OVERALL

	PILOT						
CONFIGURATION	A	B	C	D	E	F	G
B	3	3	3	2	3,3	2	
1	$2\frac{1}{2}$	3				3	3
2	3	3					6
3	3	6					2
4	4	2	4				5
5	6,9*	6					3
6	2	6					3
7	5	5					2
8		2	2				2
9	3						5
10	$2\frac{1}{2}$						3
11	7					8	2
12	9						6
13						$5\frac{1}{2}$	9
14	4						5
16				2			
17	2	3	2	4	2		
17 + L/L	4	3		3	4		
18	3						
19	$2\frac{1}{2}$						
20	2	5		4	3		
21	4	6	3			5	
22	7	9	5			6	
23	4	3	5				
24	4	3	5		3	$3\frac{1}{2}$	
25	3,4	5	8	4		7	
26	$2\frac{1}{2}$	4	4				
27	5	8	6		8	$4\frac{1}{2}$	
28	3,8	10	7	8		5	
22A	$4\frac{1}{2}$	9					
25A	4	5					
28A	5	10					

* had +200 ms delay inserted inadvertantly

Table 12
PIO RATINGS

	PILOT						
CONFIGURATION	A	B	C	D	E	F	G
B	1	1	1	1	1,1	1	
1	1	1				1	1
2	1	1					1
3	1	1					1
4	1	1	2				1
5	3,6*	3					1
6	1	1					1
7	1	1					1
8		1	1				1
9	1						1
10	1						1
11	4					4	2
12	5						3
13						1	5
14	3						1
16				1			
17	1	1	1	1	1		
17 + L/L	1	1		1	1		
18	1						
19	1						
20	1	1		1	1		
21	4	4	1			3	
22	4	6	2			4	
23	1	1	1				
24	1	1	1		1	1	
25	1,1	3	3	1		4	
26	1	1	1				
27	4	5	4		5	4	
28	2,6	6	4	3		3	
22A	3	5					
25A	1	1					
28A	4	6					

* had +200 ms delay inserted inadvertantly

4.4 PILOT COMMENTS

The reader is referred to Appendix D which presents the complete transcribed evaluation pilot comments for each evaluation. Included in these comments are some pertinent post flight comments as well as the in flight comments.

4.5 APPROACH TIME HISTORIES

Time histories of the approaches are presented in Appendix E. These time histories cover the final twenty seconds of an approach and show control activity and important responses of the aircraft.

Section 5

INTERPRETATION OF THE RESULTS USING MIL-F-8785(C) AS A FLIGHT CONTROL DESIGN CRITERIA

5.1 INTRODUCTION TO ANALYSIS

The results of the experimental flight test program strongly indicate that the response of primary interest to a flight control system designer is angle of attack. Angle of attack is the fundamental and therefore, the most critical physical quantity associated with the maneuvering flight of an airplane. The purpose of an elevator on an airplane is to rotate the airplane with respect to the relative wind, therefore changing the angle of attack. The resulting change in lift enables the airplane to maneuver, thereby altering its flight path.

A pilot, however, does not normally sense angle of attack and senses changes in flight path angle often only through vertical acceleration as a surrogate variable or perhaps directly when close to the ground. The variable the pilot senses most directly and consciously during maneuvers appears to be attitude changes.

Considerable flight experimentation results suggest that the primary response of interest to a pilot is flight path; i.e., the pilot almost always manipulates his controls in order to change the direction of the flight of the airplane. The variable he senses most often, however, appears to be changes in attitude. It can be concluded, and considerable evidence exists to support the contention that the pilot wants the airplane to fly in a direction proportional to the direction being pointed. The key to whether or not changes in attitude can be used as a surrogate or substitute for changes in flight path angle is the behavior of the angle of attack response of the vehicle (or changes in lift if other than an elevator alone is used for maneuvering purposes).

The results of the flight experiments reported upon in this document suggest that a pilot prefers a flight control system designed such that he can use changes in pitch angle as a surrogate for changes in flight path angle. If the correspondence between pitch angle and flight path angle is inadequate, however, there are indications that he will judge the vehicle flying qualities directly on the basis of the $\dot{\gamma}$ (and/or γ) dynamic behavior of the vehicle.

Flying qualities requirements, as defined by MIL-F-8785(C) concentrate on the quality of the angle of attack response of the vehicle. The data used to formulate the specification was obtained primarily from conventional or angle of attack command airplanes. Because (to a fairly good approximation for conventional aircraft), $\dot{\gamma} \approx Z_{\alpha}\alpha$ and because for a majority of the flight experiments the pilot was located near the center of rotation (percussion), the specification on angle of attack served equally well for $\dot{\gamma}$ or n_z, as well. A conventional or angle of attack command airplane has an $\alpha/\delta(s)$ transfer function characterized usually by phugoid poles placed relatively close to the lower frequency zeros, therefore, there is little residue of the phugoid mode in the angle of attack response. In the long term response of the vehicle there was little question of the use of pitch angle as a surrogate for flight path; since $\dot{\alpha}(t) \doteq 0$ after the short term response, $\Delta\gamma \doteq \Delta\theta$ and changes in pitch attitude indicated corresponding changes in flight path angle.

The ω_n vs n/α (constant CAP) requirement in MIL-F-8785(C) defines the region of acceptable short period frequency as a function of n/α and therefore, with the addition of an acceptable ζ_{sp} range, define the ranges of acceptable $\alpha(t)$ dynamic behavior of the vehicle in the short term. The associated acceptance correlation parameter between angle of attack (or in conventional aircraft, n_z) and pitching motions is defined by $CAP = \dfrac{\ddot{\theta}|_{t=0^+}}{n_z|_{ss_{sp}}}$. Alternately, the CAP parameter is defined as $CAP \doteq \dfrac{\omega_n^2}{n/\alpha} = \dfrac{\omega_n^2}{V/g(1/\tau_{\theta_2})}$. For an acceptable angle of attack (or $n_{z_{sp}}$) response, the dynamics of the pitch attitude response is defined as a function of velocity and $1/\tau_{\theta_2}$. Therefore, the ω_n vs n/α item in MIL-F-8785 (C) describes not only the acceptability of the angle of attack behavior of the airplane, but also defines the harmony between rotation and translation (pitch and heave) in the short term.

In the long term, the experimental results reported upon herein strongly suggest a requirement that in the long term the parameter of importance is correspondence between changes in γ and θ. The smaller the phugoid or low frequency residue in $\alpha(t)$ the more likely will be the acceptance by the pilot of the dynamics of the vehicle.

It can be shown that many of the recently proposed flying qualities indicators correlate with the requirements on $\alpha(t)$. For instance, the pilot dislikes pitch "bobble" unless γ also "bobbles" in adequate correspondence. The pilot generally does not prefer a high phugoid frequency except if the vehicle is α command, for then $\dot{\alpha}(t) \cong 0$ in the long term and $\Delta\theta = \Delta\gamma$. If the control system is configured as a rate command system in the short term (i.e., no pitch "overshoot") then $1/\tau_{\theta_2}$ should have a value sufficiently large such that the angle of attack response, (which is dominated by $1/\tau_{\theta_2}$ in this type of system), is sufficiently rapid to satisfy the ω_n vs n/α requirement of MIL-F-8785(C). Finally, if the phugoid frequency is very low (or approximately equal to $1/\tau_{\theta_1}$) then the long term residue in both α and q is small and the flying qualities are enhanced. In fact, if the long term modes of motion were entirely decoupled from pitch and heave, it appears to make little difference whether the flight control system is configured as an angle of attack or pitch rate command system, and in fact the pitch rate command system appears to be preferred because the pilot has confidence that flight path will be in correspondence with the pitch commands.

5.2 ANALYSIS OF THE RESULTS

5.2.1 Comparison with ω_n vs n/α Requirements

Table 13 summarizes the pilot ratings for the first 14 dynamic configurations of the experimental flight test program. Separate evaluations were solicited for the approach and the flare/touchdown segments of the landing task.

Figure 18 locates the configurations with respect to the ω_n vs n/α constant CAP requirement of MIL-F-8785(C) of the first 14 configurations. Configurations 2, 4 and 10 are predicted to be Level 2, because the damping ratio was higher than the allowable $\zeta = 1.3$, while Configuration 11 and 12 are not included because they are higher order. MIL-F-8785(C) accurately predicts the average ratings of 7 of the 12 configurations for the approach, and only 4 of the 12 configurations during flare and landing. The rather poor predictive ability of MIL-F-8785(C) strongly suggests the need for criteria improvements relating to piloting technique and/or vehicle dynamics command configurations. Configurations 13 and 14 indicate strongly that the flare and landing task may be quite demanding, requiring considerably more stringent constraints than those outlined in the Category C requirements of MIL-F-8785(C). As shown in Figure 18, Configurations 13 and 14 would be rated Level 1 according to the Category C requirements and for the approach phase of flight, the C requirement accurately predicts the rating. For flare and landing, the ratings were Level 2 and 3, and as shown in the figure, more accurately corresponding to the flight precision requirements of Category A flight.

Table 13

PILOT RATINGS SUMMARY

Configs.	Approach		Flare and Land		Predicted FQ Level (MIL-F-8785-C)
	Actual	Mean	Actual	Mean	
1	2,3,3,3	2.8	$2\frac{1}{2}$,3,3,3	2.8	1
2	2,3,3	2.7	3,3,6	4.0	2
3	3,6,2	3.7	3,6,2	3.7	1
4	4,2,4,5	3.8	2,2,4,5	3.2	2
5	4,5,3	4.0	6,6,3	5.0	1
6	2,6,3	3.7	2,3,3	2.7	1
7	4,4,2	3.3	5,5,2	4.0	1
8	3,2,2	2.3	2,2,2	2.0	1
9	3,4	3.5	3,5	4.0	1
10	$2\frac{1}{2}$,2	2.3	$2\frac{1}{2}$,3	2.8	2
11	5,5,2	4.0	7,8,2	5.7	
12	6,4	5.0	9,6	7.5	
13	3,3	3.0	$5\frac{1}{2}$,9	7.3	1
14	3,3	3.0	4,5	4.5	1

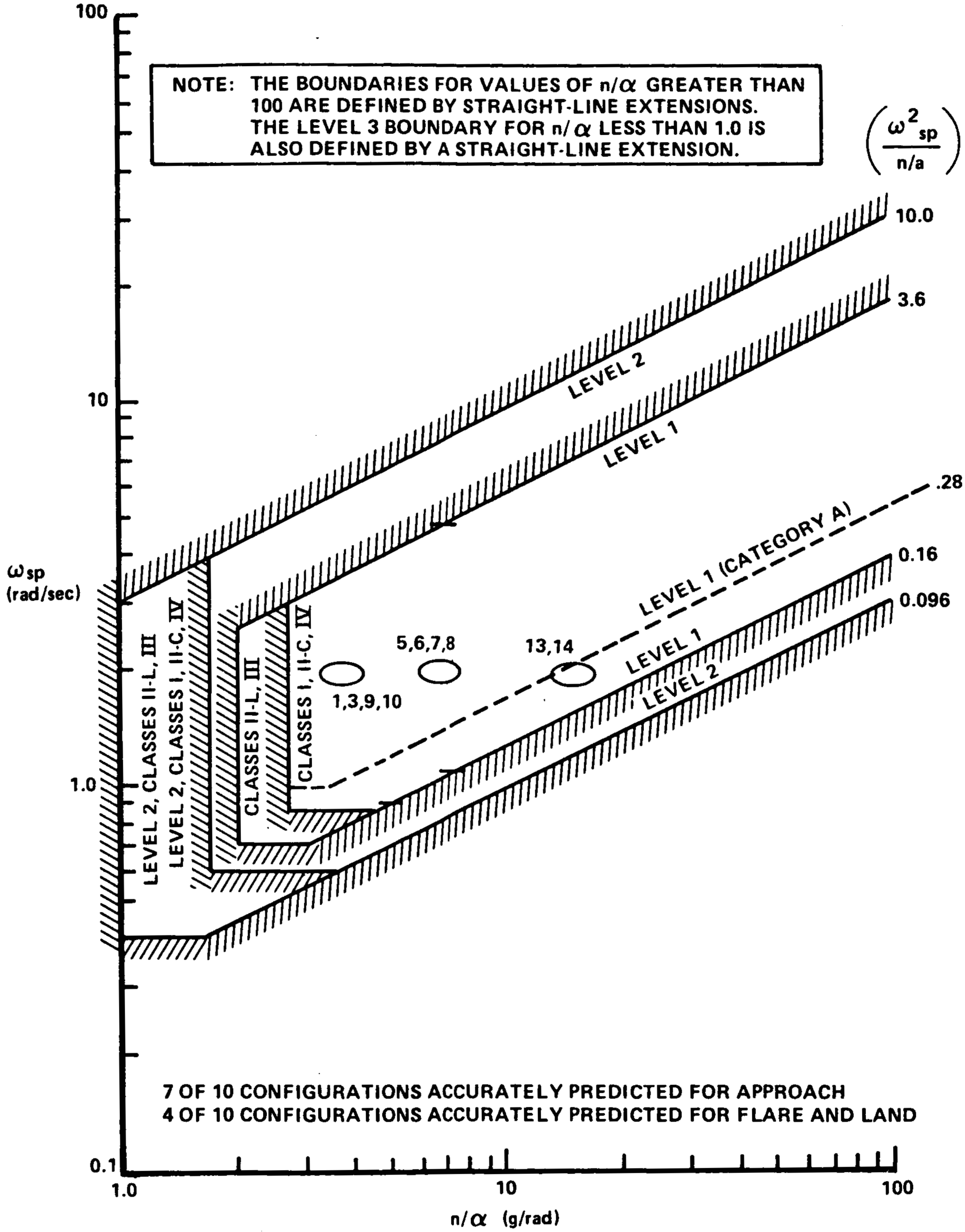


Figure 18. COMPARISON WITH MIL-F-8785(C) SHORT PERIOD REQUIREMENTS
7 OF 12 CONFIGURATIONS CORRECTLY PREDICTED

5.2.2 Comparison with $\omega_n \tau_{\theta_2}$ vs ζ Requirements

Figure 19 locates the configuration with respect to the $\omega_n \tau_{\theta_2}$ vs ζ_{sp} requirements of the proposed MIL-F-8785(C) Standard and Handbook. This figure accurately predicts the mean flying qualities level for only 4 of the 12 configurations for the flare and landing task whether it is considered to be Category C or if the task is considered to be Category A.

5.2.3 Time Domain Analysis

An alternate way to interpret the ω_n vs n/α requirement of MIL-F-8785(C) is to develop a normalized time history response envelope of $\alpha(t)$ based upon the range of allowable short period frequency and damping ratio for a particular value of n/α. Because, however, there is some question as to whether the most appropriate requirement is defined by Category A or Category C precision as applied to the flare and landing task, an angle of attack time history envelope was developed directly from the flight data found in Reference 17, the data base used to formulate the MIL-F-8785(B) requirements. The development of this time history envelope, shown in Figure 20 is detailed in Reference 11. The data used to develop the time history envelope covered a range of n/α of 3.5 to 11.8, so the envelope would be expected to be less broad (i.e., more stringent) than an envelope obtained by transformation of the ω_n vs n/α requirements of MIL-F-8785(C) for each value of n/α.

The flight data from which the $\alpha(t)$ envelope of Figure 20 was developed has two major limitations or restrictions. First, the data were collected from variable stability aircraft that were mainly angle of attack command or conventional aircraft configurations in that the phugoid mode was normally lightly damped but dynamically stable. The second limitation is that the phugoid dynamics were not specifically documented with the data, so the effects of the phugoid mode residues could not be included - the $\alpha(t)$ time history response envelope considered only the part of the angle of attack response attributable specifically to the short period mode. Because of these restrictions it is expected that only those configurations that were angle of attack command in the long term, i.e., $\alpha(t)$ approached a constant value as $t \to \infty$ could be evaluated with respect to the envelope. Therefore, Configurations 2, 3, 6, 7, 10, 11, 12, and 14 were not evaluated with respect

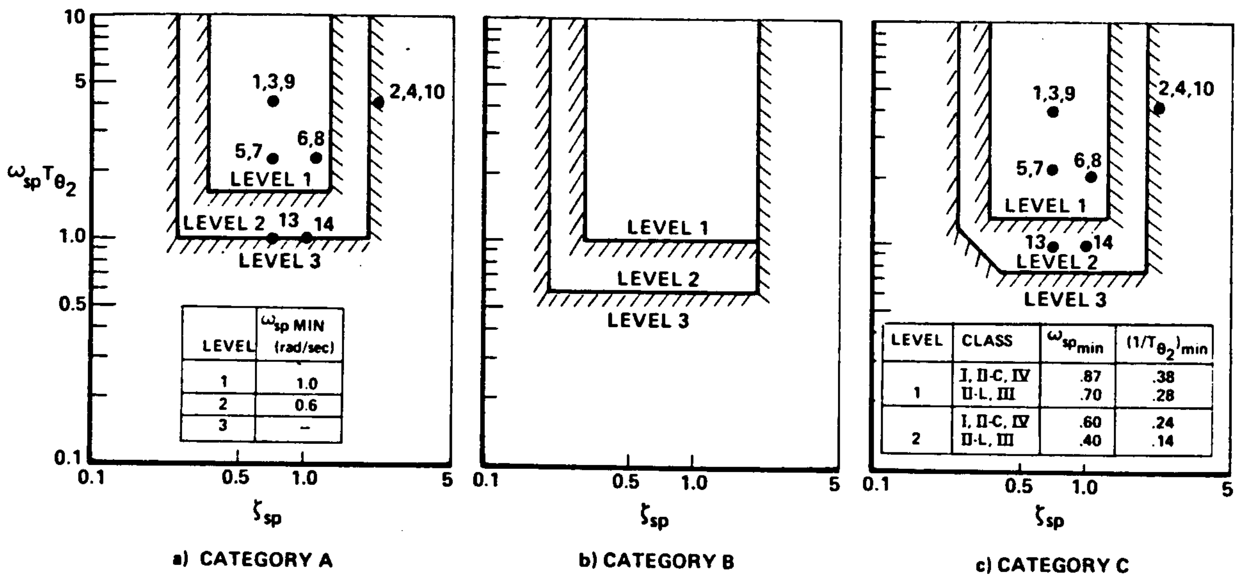


Figure 19. CONFIGURATIONS 1-10, 13 AND 14 $\omega_n\tau_{\theta_2}$ vs. ζ_{sp} REQUIREMENTS
4 OF 12 CONFIGURATIONS CORRECTLY PREDICTED

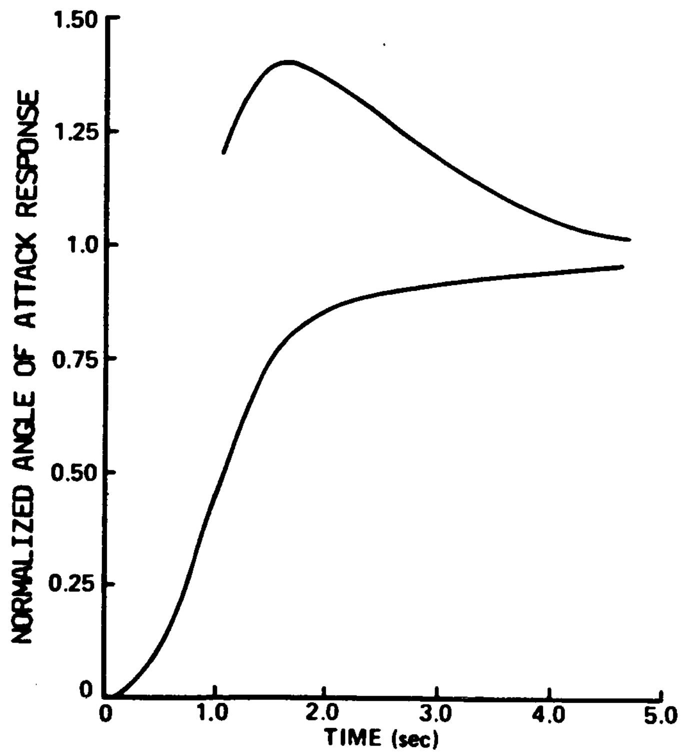


Figure 20. TIME HISTORY RESPONSE ENVELOPE

to the time history response envelope. The responses of Configurations 1, 4, 5, 8, 9, 13, and 16 are shown in Figures 21 through 26 with respect to the time history envelope criteria. The results indicate that the angle of attack time history response envelope criteria is as accurate an indicator of Level 1 flying qualities as the ω_n vs. n/α requirement of MIL-F-8785(C). In fact, the $\alpha(t)$ envelope, with the restriction that the responses represent only the short period mode, is a direct time domain representation of the ω_n vs. n/α requirement.

It should be stressed that no single flying qualities indicator is fool-proof in the sense that exceptions do not exist. The purpose of an indicator such as the angle of attack time history envelope is to serve as a guide to the flight control system designer. The satisfaction of an indicator merely serves to increase the probability that his design will have Level 1 flying qualities. Many other factors, including individual piloting technique have been shown to affect the pilot rating of a configuration. Agreement with indicators, however, tend to account for a wide range of piloting technique because the flight experiments flown to form a data base were performed with a variety of pilots and their associated varying techniques. In these experiments, Pilot G exhibited a smooth technique with minimum stick activity while Pilot B was constantly pumping or "dithering" the stick. As is to be expected, Pilot G tended to prefer the angle of attack command configurations while Pilot B tended to prefer the pitch rate command configurations in which a pulsing stick input command technique produced approximately the same type of response as the smoother step command inputs of the angle of attack command configurations. It appears, however, that if the angle of attack response is well behaved as defined by the angle of attack response envelope, both piloting techniques can be accomodated. As seen from the figures, Configuration 1, an angle of attack command configuration in both the short and long term and Configuration 8, a pitch rate command configuration in which the angle of attack response almost satisfies the criterion, were both rated Level 1 by all three pilots who evaluated the configuration.

One significant exception to the angle of attack response envelope rule is noted. Configuration 10, with no phugoid residue in either angle of attack or pitch rate was rated Level 1 although the angle of attack response did not satisfy the angle of attack response envelope criteria.

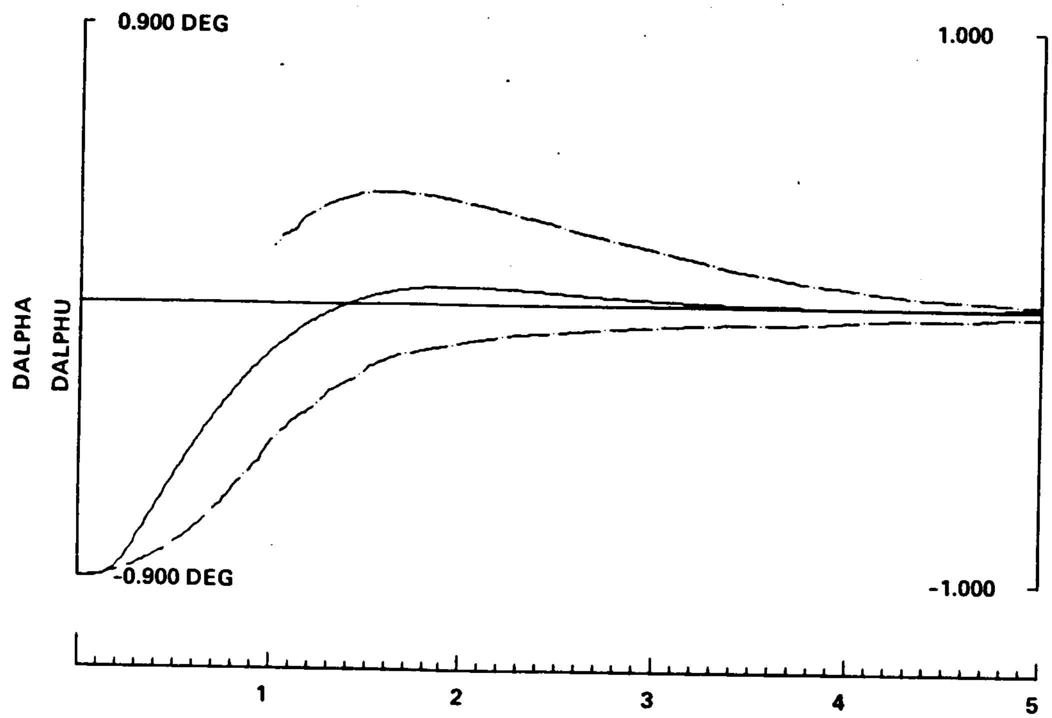


Figure 21. NORMALIZED ANGLE OF ATTACK RESPONSE CONFIGURATION 1

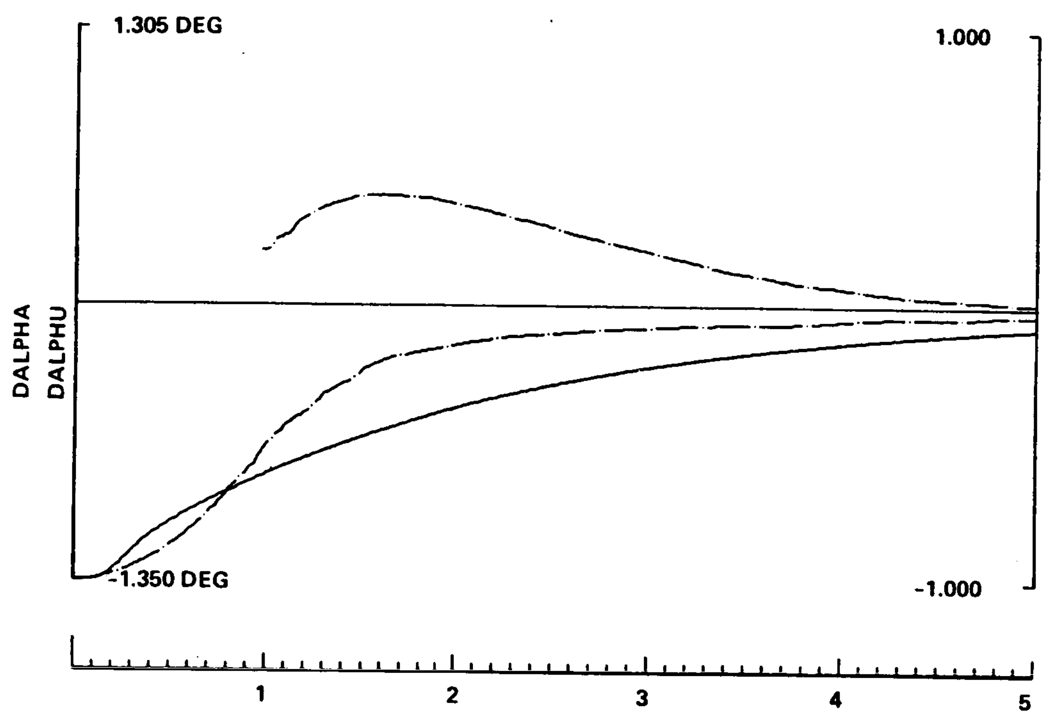


Figure 22. NORMALIZED ANGLE OF ATTACK RESPONSE CONFIGURATION 4

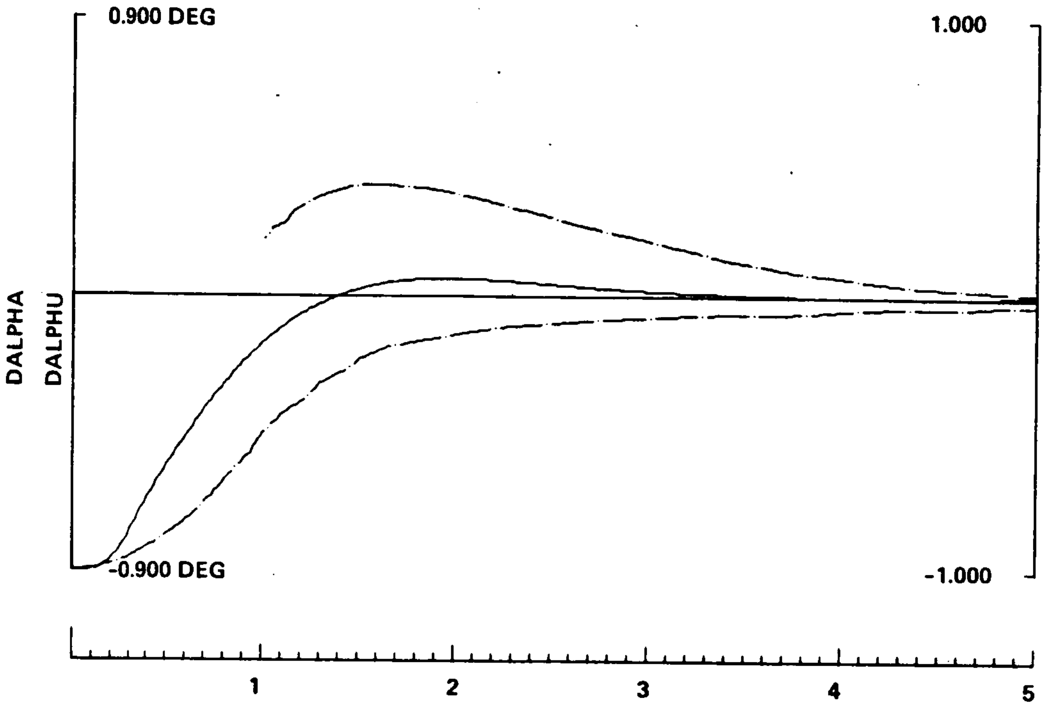


Figure 23. NORMALIZED ANGLE OF ATTACK RESPONSE CONFIGURATION 5

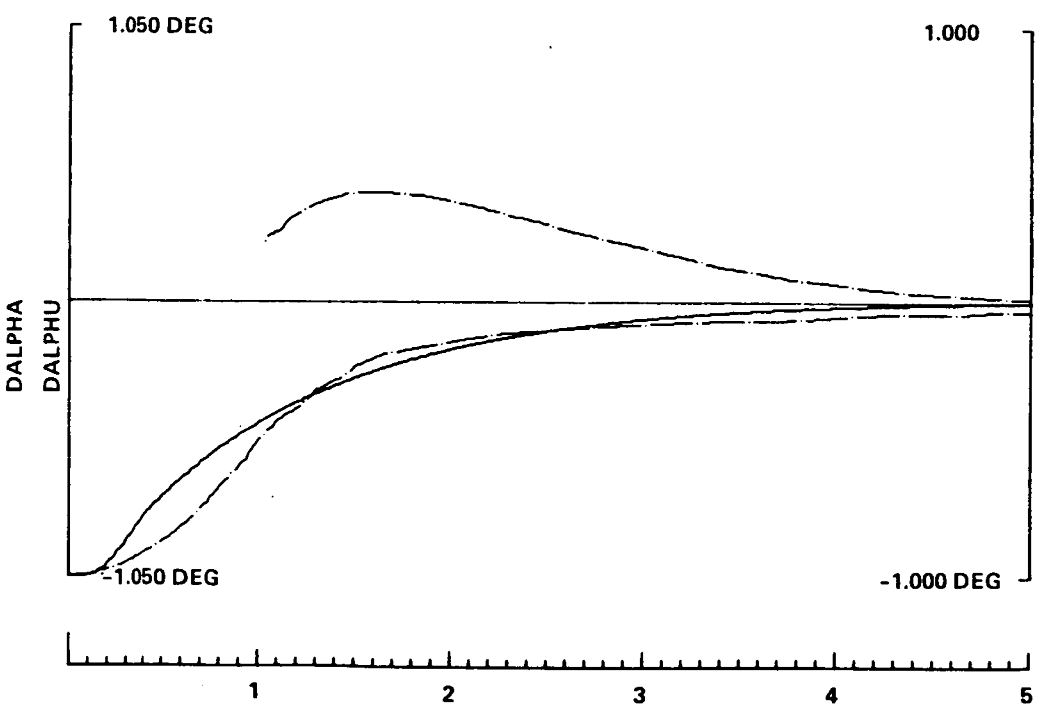


Figure 24. NORMALIZED ANGLE OF ATTACK RESPONSE CONFIGURATION 8

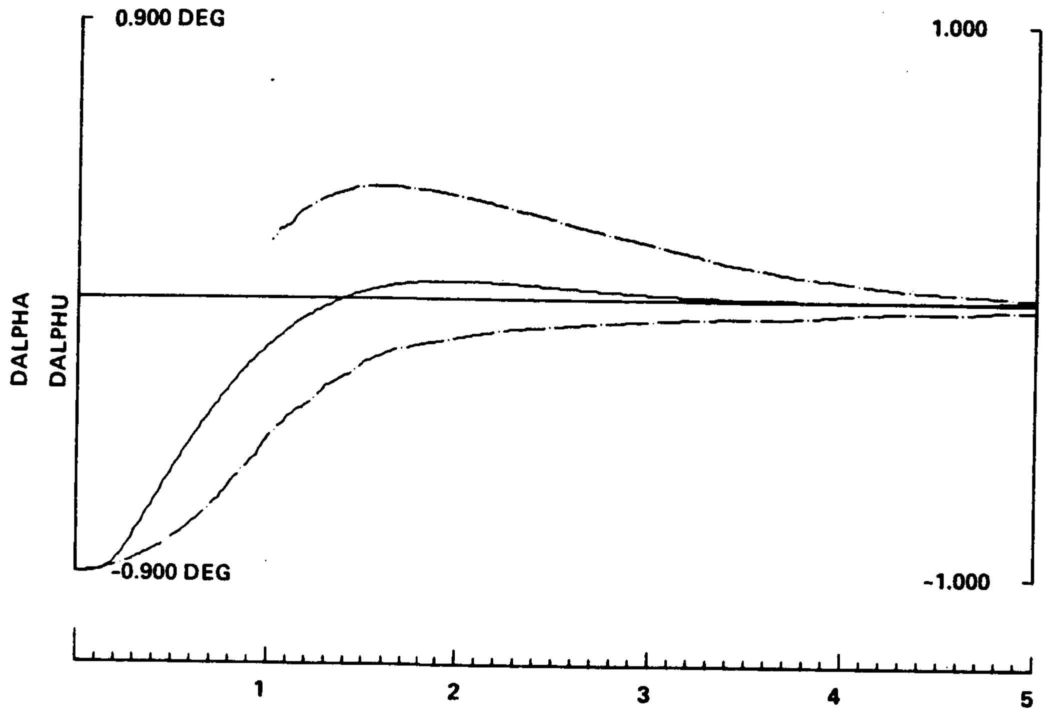


Figure 25. NORMALIZED ANGLE OF ATTACK RESPONSE
CONFIGURATION 9, 13

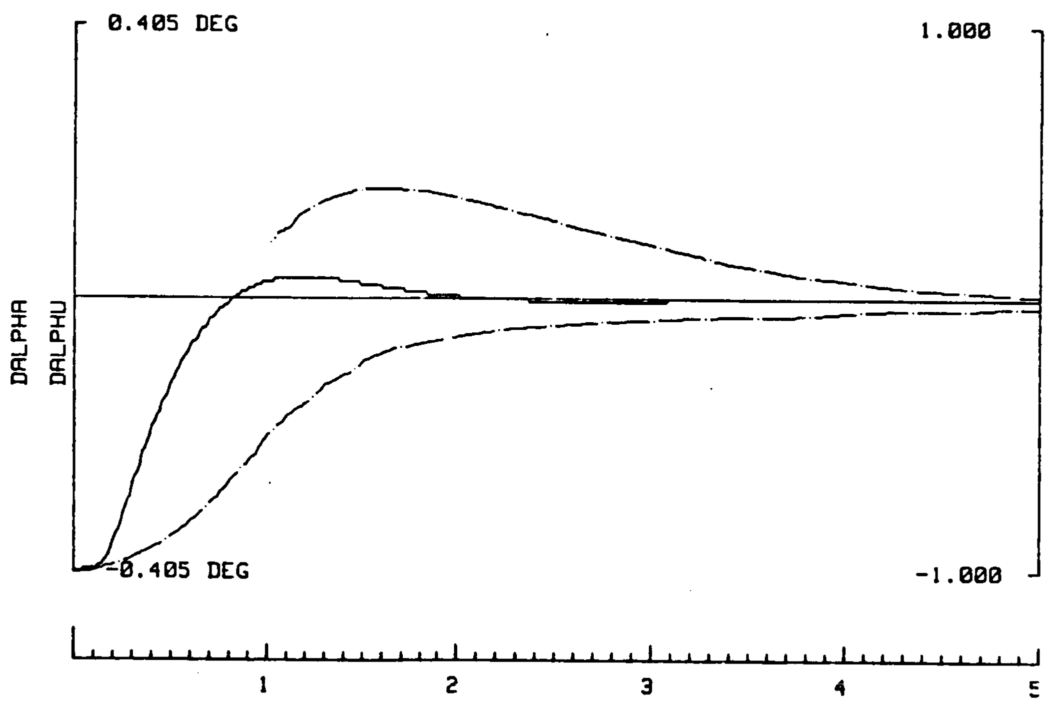


Figure 26. NORMALIZED ANGLE OF ATTACK RESPONSE
CONFIGURATION 16

The lessons learned by the analysis shown in this section is that the flying qualities are most likely to be rated acceptable by a range of pilots with differing piloting techniques if:

1. The angle of attack response complies well with the angle of attack response envelope and
2. The system is configured as angle of attack command in the long term or phugoid mode.

It is difficult to express the angle of attack response requirements in terms of the other responses of the vehicle. For instance, Configurations 11 and 12 showed smooth, well behaved short term responses in both pitch rate and normal accelerations, and suggest compliance with the short term requirements. The lack of harmony between attitude and flight path, however, is demonstrated by the fact that the angle of attack response is not well behaved. An equivalent system analysis of angle of attack yielded results showing angle of attack to be approximated accurately by an integrator, K/s, and has poor flying qualities.

It appears possible to formulate an angle of attack response time history envelope to include the effects of phugoid mode residue. Indications are that some residue is allowable, but not very much. It is suggested that a Level 1 criteria could allow a variation from the short-period-only steady state value of approximately +10%, -5% during the first 5 or 6 seconds of the angle of attack response to a pilot step stick command input.

Phugoid or Long Term Dynamics

Perhaps more clearly than any other result, the preference is that the airplane fly in the direction it is pointing, i.e., in the long term the preference is for $\Delta\theta \cong \Delta\gamma$. This observation seems to be much more important than specific values of phugoid frequency and damping ratio. Configuration 1, an angle of attack command aircraft, shows large phugoid residue in both pitch angle and flight path angle, but the two angles are in harmony, i.e., $\Delta\theta = \Delta\gamma$ because $\dot{\alpha}(t) \doteq 0$ in the long term. The configuration is Level 1. Configurations 11 and 12, the $\dot{\gamma}$ command configurations, show large differences between changes in attitude and changes in flight path angle and are rated Level 2 and

Level 3. The only difference between Configurations 6 and 8 is the better harmony between θ and γ for Configuration 8, which is angle of attack command in the long term. Configuration 8 was rated Level 1 by all the pilots. One can easily conclude that the most important parameter in the long term for manual flight in the flare and landing is the harmony between attitude and flight path angle. Therefore, the most important criteria for long term behavior is the difference between θ and γ, which is angle of attack.

The results of this experimental flight test program suggest that the flight control system should be designed to be angle of attack command in the long term regardless of the short term command configuration. If attitude "hold" is desirable, then the zeros of the angle of attack transfer function should be approximately the same as the zeros of the pitch rate transfer function, thereby minimizing phugoid mode residue in both the angle of attack and pitch rate. To do this would require a second controller, such as a canard, used independently of the elevator. Configuration 10, designed as a pitch rate command system with no phugoid residue in either α or $\dot{\theta}$ was clearly rated Level 1 by both pilots for flare and landing even though $1/\tau_{\theta_2}$, which dominates the $\alpha(t)$ response, would be considered too low for a Level 1 rating. This might be interpreted to mean that a small increase in γ after θ has been positioned by the pilot is acceptable. Configuration 2, which was not rated as highly but had the same short period dynamics as Configuration 10, showed a decrease or "droop" in γ after the attitude was set by the pilot. In rate command systems, attitude might be used by the pilot as an anticipatory cue that γ will follow. When this does not happen, a larger value of $1/\tau_{\theta_2}$ appears to be required to show the θ-γ relationship more quickly to the pilot.

It appears then that if the designer were to try to obtain optimum flying qualities for a rate command/attitude hold system, he might require either a canard with low control power to minimize the angle of attack residue in the long term, or a direct lift flap with high control power to optimize the effective value of $1/\tau_{\theta_2}$.

5.2.4 Design of Rate Command Systems

One of the major objectives of the approach and landing flying qualities experimental program was to try to transform the flying qualities requirements into a format more compatible with the techniques of the flight control system designer. Although the experimental program did not include higher order systems such as would be obtained if a proportional plus integral compensation network were used in the design process, the experiment did yield valuable information for lower order systems, such as would be attained if pole placement or model following methods were used in the design process. Because pole placement methods, using either state feedback or deterministic observer techniques, do not increase the order of the system response, the experiment was designed to define flying qualities/flight control system requirements under conditions that would be expected to yield optimum flying qualities.

For a rate command system, one of the short period poles is placed at the zero of the constant speed, two degree of freedom $q/F_S(s)$ transfer functions, i.e., at $s = -1/\tau_{\theta_2}$. The other pole is defined from the relationship $\tau_{\theta_2}\omega_n^2 = p$. Because the maximum short period damping ratio is specified to be $\zeta_{sp_{max}} = 1.30$, a relationship can be obtained to directly relate the minimum value of $1/\tau_{\theta_2}$ and short period frequency to obtain Level 1 flying qualities as defined by MIL-F-8785(C). From the transfer function

$$\frac{\alpha}{F_s}(s) = \frac{K}{s^2 + 2\zeta\omega_{sp}s + \omega_{sp}^2} \tag{1}$$

the two relationships are obtained

$$\omega_{sp}^2 = p(1/\tau_{\theta_2}) \tag{2}$$

$$2\zeta_{sp}\omega_{sp} = p + 1/\tau_{\theta_2} \tag{3}$$

With a maximum short period damping ratio of $\zeta_{sp} = 1.30$, the relationship between short period frequency and $1/\tau_{\theta_2}$ for Level 1 flying qualities is therefore defined by MIL-F-8785(C) to be

$$2.131\ \omega_{sp} \geq 1/\tau_{\theta_2} \geq .469\ \omega_{sp} \tag{4}$$

The above equation specifies a required range of $1/\tau_{\theta_2}$ as a function of short period frequency. For the example of this experiment, $\omega_{sp} = 2.0$ with a value of $1/\tau_{\theta_2} = 0.5$, such as defined for the rate command Configuration 2, would not satisfy the above relationship and could not be expected to be rated Level 1 on a consistent basis. For this case, the angle of attack command system of Configuration 1 was indicated, and was shown to be Level 1. The rate command Configuration 6 was chosen with $\omega_{sp} = 2.0$, $1/\tau_{\theta_2} = 0.9$, a value close to the minimum $1/\tau_{\theta_2}$ requirement of,

$$1/\tau_{\theta_2} = .469\ \omega_{sp} \tag{5}$$

and was given improved pilot ratings. The relationship indicated by Equation (4) is depicted graphically in Figure 27. The two lines drawn in the figure show the two equality relationships of Equation (4). The angle of attack/pitch rate preference boundary therefore represents an alternate way to define a maximum damping ratio of $\zeta = 1.3$ specified in MIL-F-8785(C).

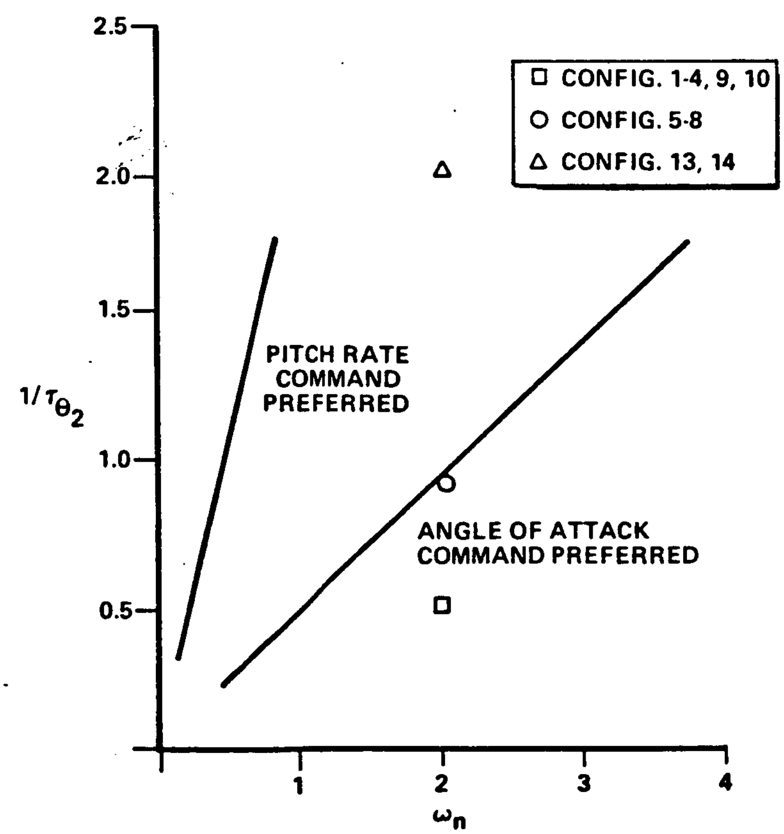


Figure 27. SHORT TERM PREFERENCE AS SPECIFIED BY MIL-F-8785(C)

5.3 EFFECT OF PILOTING TECHNIQUE

The evaluation pilots who participated in the program had varied backgrounds. Some were past participants in similar experimental flight test programs, had complete familiarization with the Cooper-Harper Pilot Rating Scale and past experience with pitch rate command flight control systems. Others were test pilots familiar with conventional large and medium sized aircraft.

The evaluation pilots displayed what is believed to be a wide range of piloting techniques during the performance of this flight experiment. A wide range of stick activity is evident even by causual perusal of the raw data. An examination of typical flare and landing data for Configuration 1, (Figure 28, 29, and 30) show widely varying technique. Pilot G used smooth, step-like inputs (Figure 30) while Pilot B superimposed a nearly constant 1 hertz "dither" pumping action (Figure 29). Pilot A flew the airplane in a fashion between Pilots G and B in terms of stick activity while relieving the required stick force bias by triming the elevator (Figure 28).

The data tends to support the hypothesis that piloting technique is related to pilot preference for a rate command or angle of attack command configuration. The most appropriate stick command input for an angle of attack command system is a smooth, step or ramp type input as exhibited by Pilot G, who tended to prefer the short term angle of attack configurations. A comparison of ratings by Pilot G between short term α-command Configuration 1 and 3, and short term q-command Configuration 2 and 4 clearly tends to support the association between technique and command configuration. Pilot B behaved oppositely. His natural tendency toward a dithering or pulsing input led him to tend to favor the rate command configurations because his inputs were more naturally in harmony with the way a rate command system is usually flown. Pilot A, on the other hand, displayed both lower frequency step and ramp-type inputs as well as a dithering input (at lower frequency than Pilot B). He flew configurations with precision and liked almost all of the Configurations 1-7.

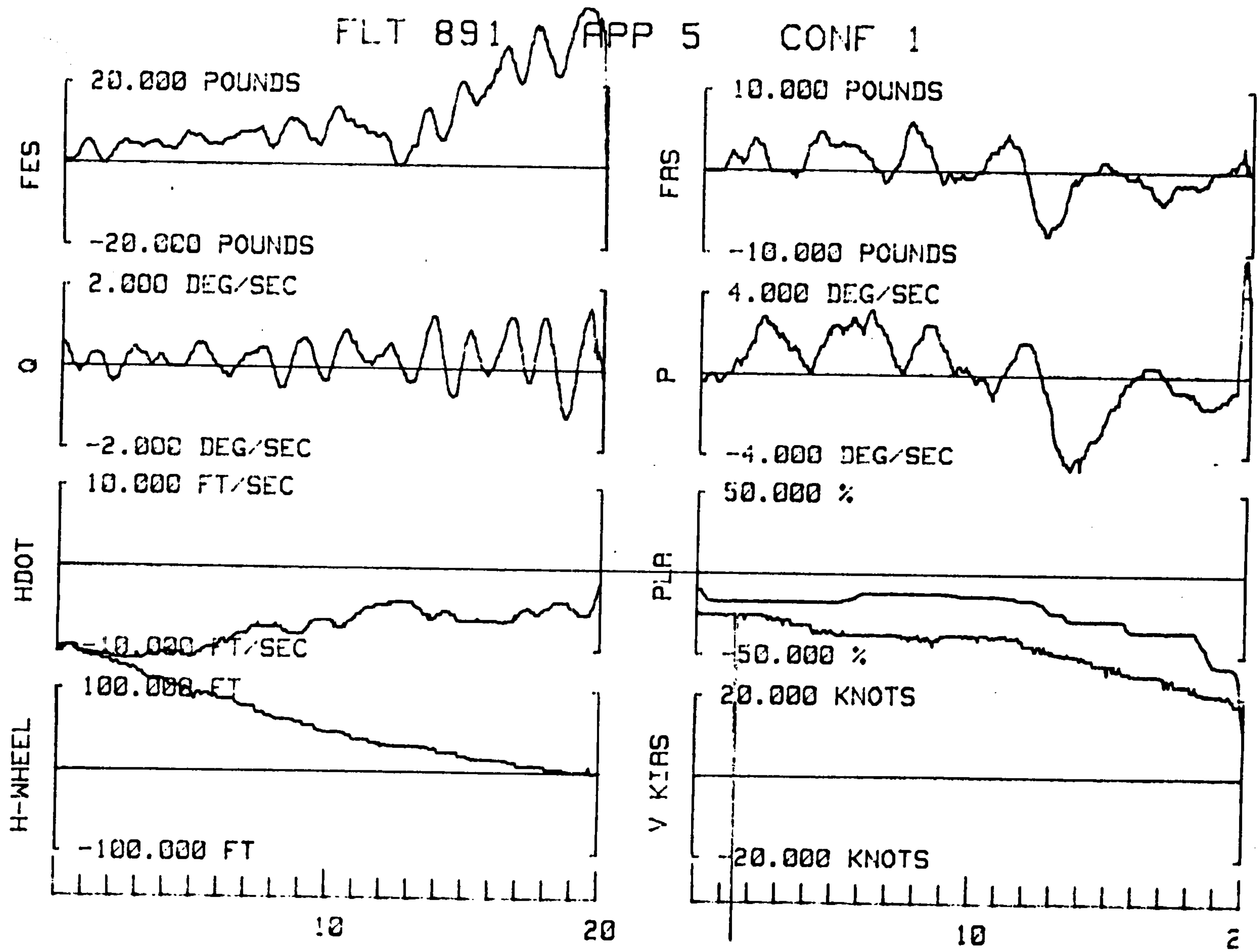


Figure 28. TYPICAL APPROACH, CONFIGURATION 1, PILOT A

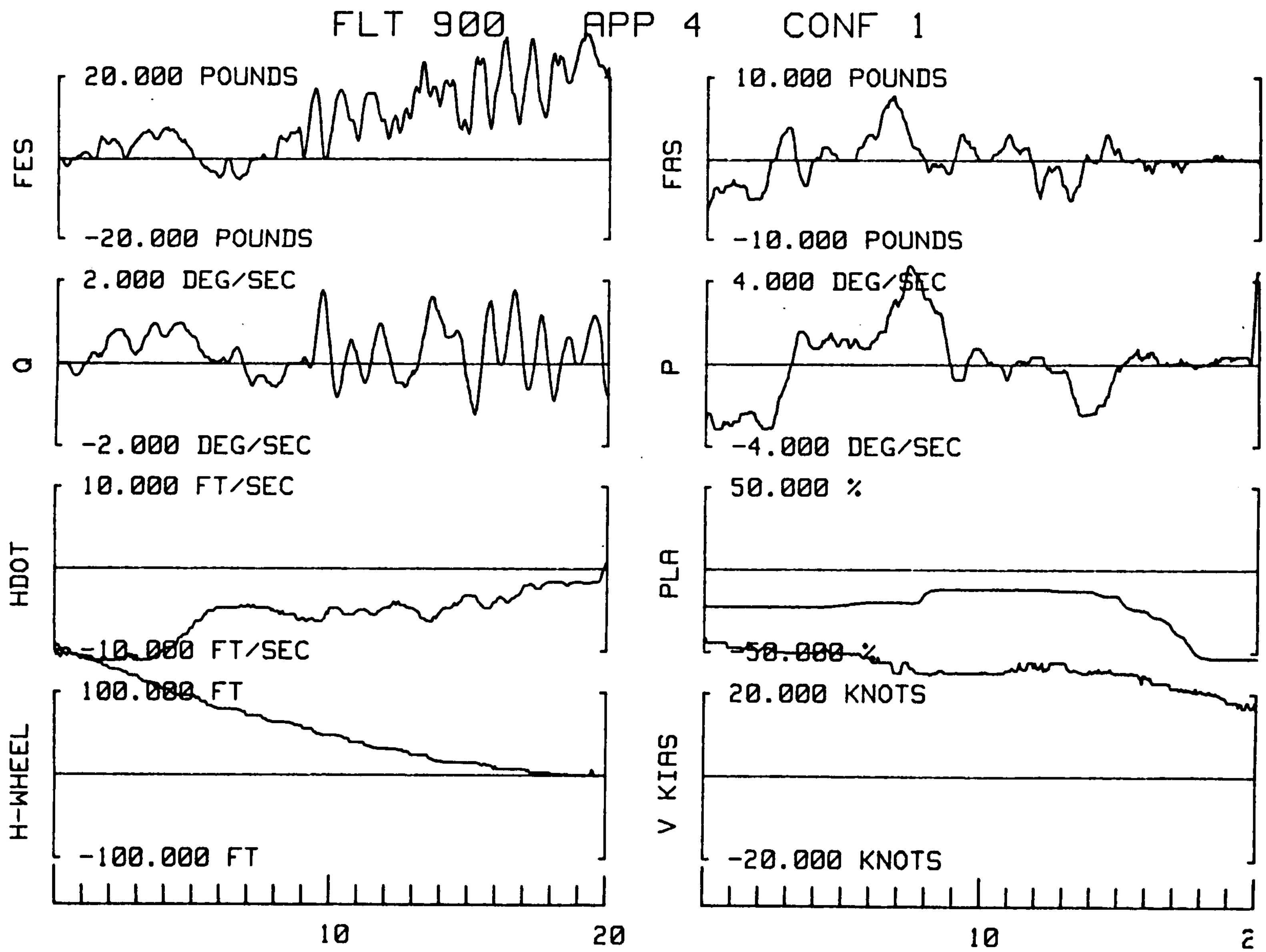


Figure 29. TYPICAL APPROACH, CONFIGURATION 1, PILOT B

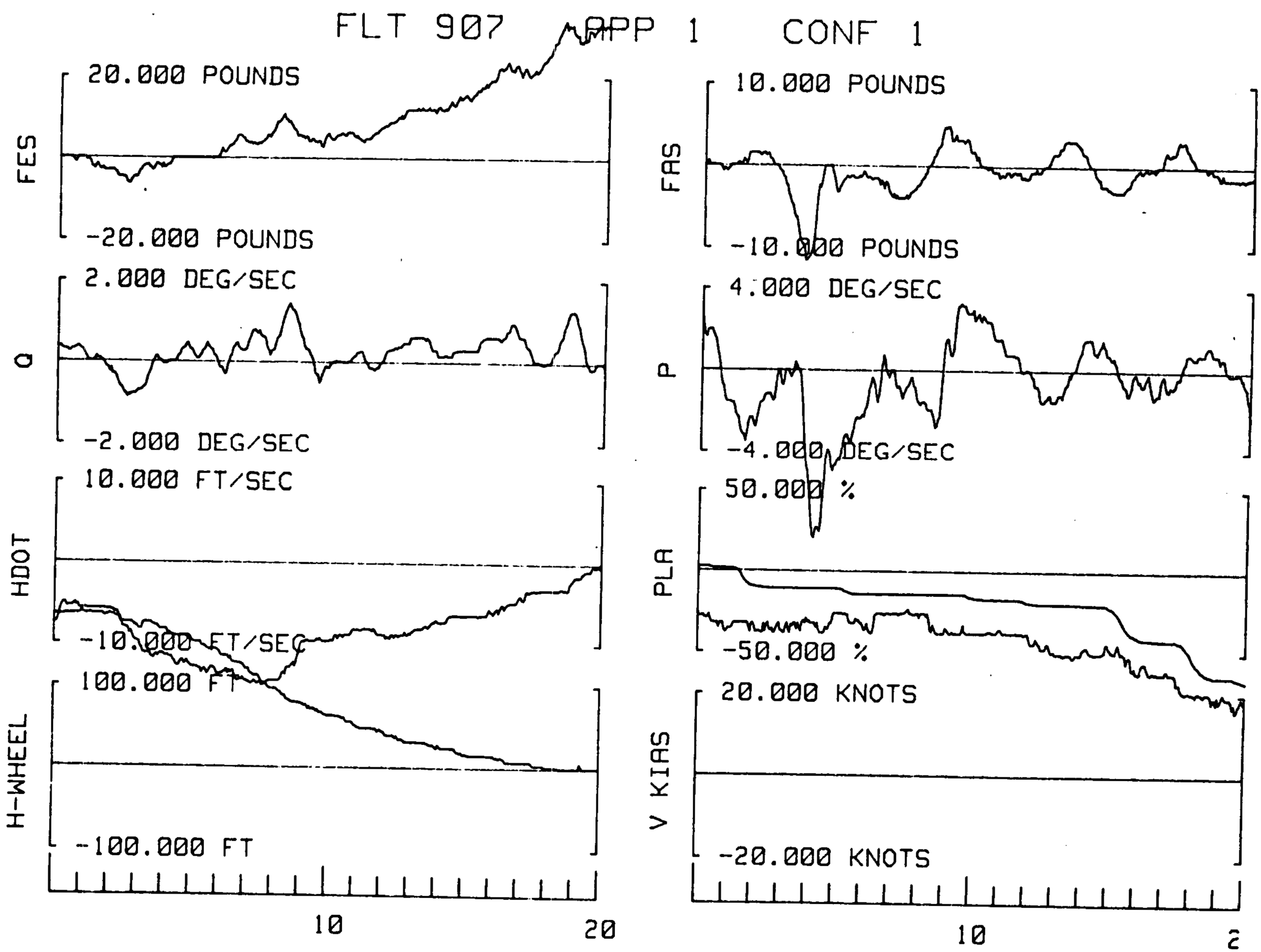


Figure 30. TYPICAL APPROACH, CONFIGURATION 1, PILOT G

Power spectral densities of pilot stick activity for the three pilots are shown in Figures 31, 32, and 33 (Reference 18). The solid line shows stick activity for the approach, while the dashed lines indicate activity during the flare and touchdown phase for Configuration 1. These plots clearly show the low frequency spectrum of Pilot G, the narrow bandwidth spectrum of Pilot B and the dual spectrum input of Pilot A.

The power spectrums change amplitude with dynamic configuration. Figure 34 and 35 show power spectrum for Pilots A and G for Configuration 14, rated Level 1 for approach but Level 2 for flare and landing. These specta show not only that the low frequency portion of the spectrum is eliminated because steady stick inputs are not required to fly the configuration, but it also shows that a pilot flys a poorly rated configuration less aggressively than one rated Level 1. The dual mode tendency of Pilot A seems to be preserved, whereas Pilot G, who was the smoothest flying pilot for Configuration 1 still flies less aggressively, choosing a low frequency input spectrum. By commanding inputs both near and above the short period natural frequency, it can be speculated that Pilot A is evaluating or anticipating both the $\dot{\theta}$ and $\dot{\gamma}$ vehicle response, while Pilot G is flying almost totally with respect to flight path.

Although no judgement can be made with respect to preferred or superior flying technique, it appears that Pilot A may be most versatile in terms of adapting to vehicle dynamic configurations because of his dual-mode stick activity spectrum tendency. The task for the flight control system designer is to try to design the system that will accomodate the widest possible range of piloting technique because it is clear that all these pilots changed their piloting technique only minimally as a function of configuration. They each seemed to evaluate the configuration with respect to their individual inherent technique. Configuration 1, a short and long term angle of attack configuration and Configuration 8, a higher $1/\tau_{\theta_2}$ short term $\dot{\theta}$ command configuration but long term α command configuration seemed to best accomodate the wide range of piloting techniques used in this program.

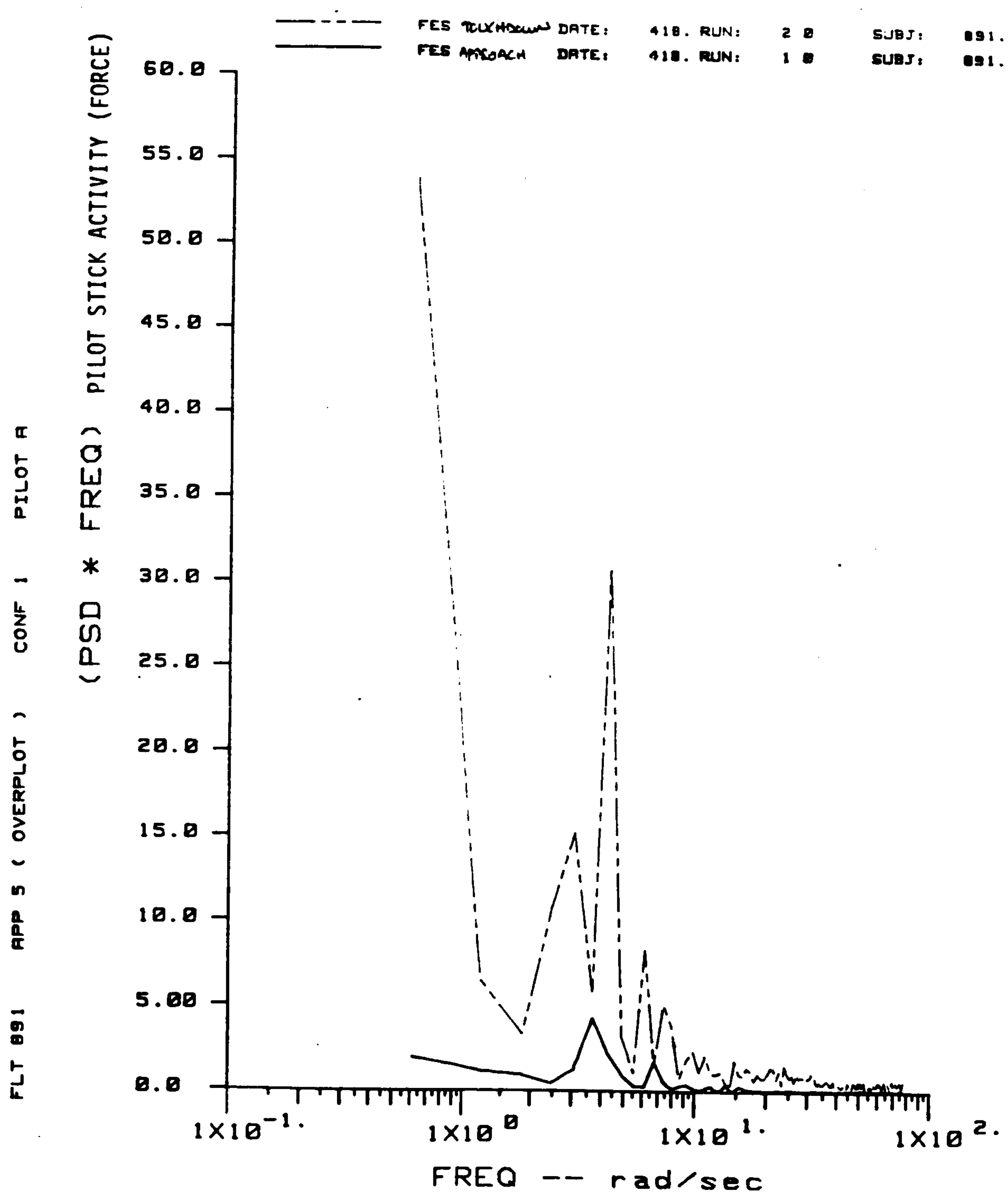


Figure 31. PSD, CONFIGURATION 1, PILOT A

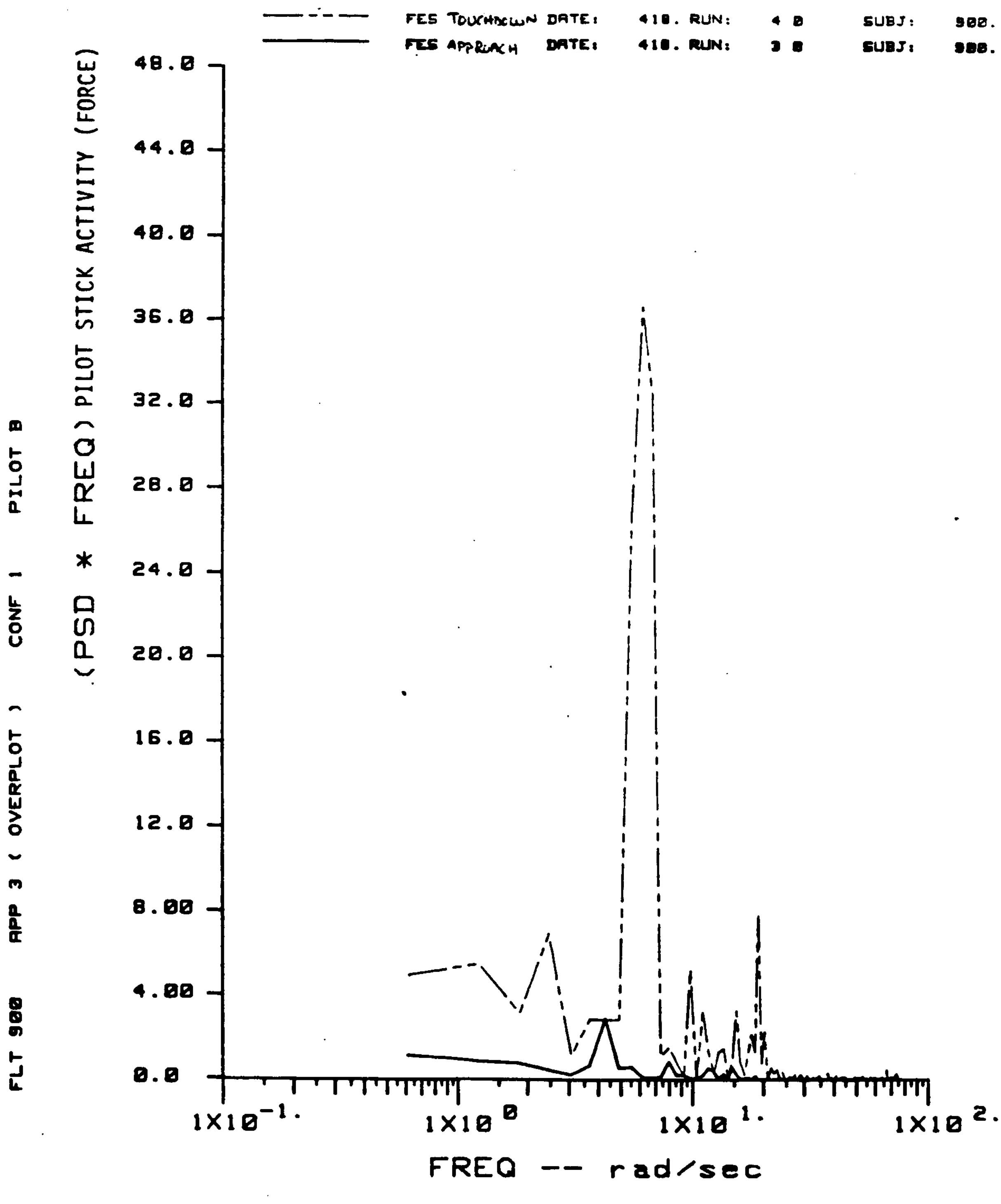


Figure 32. PSD, CONFIGURATION 1, PILOT B

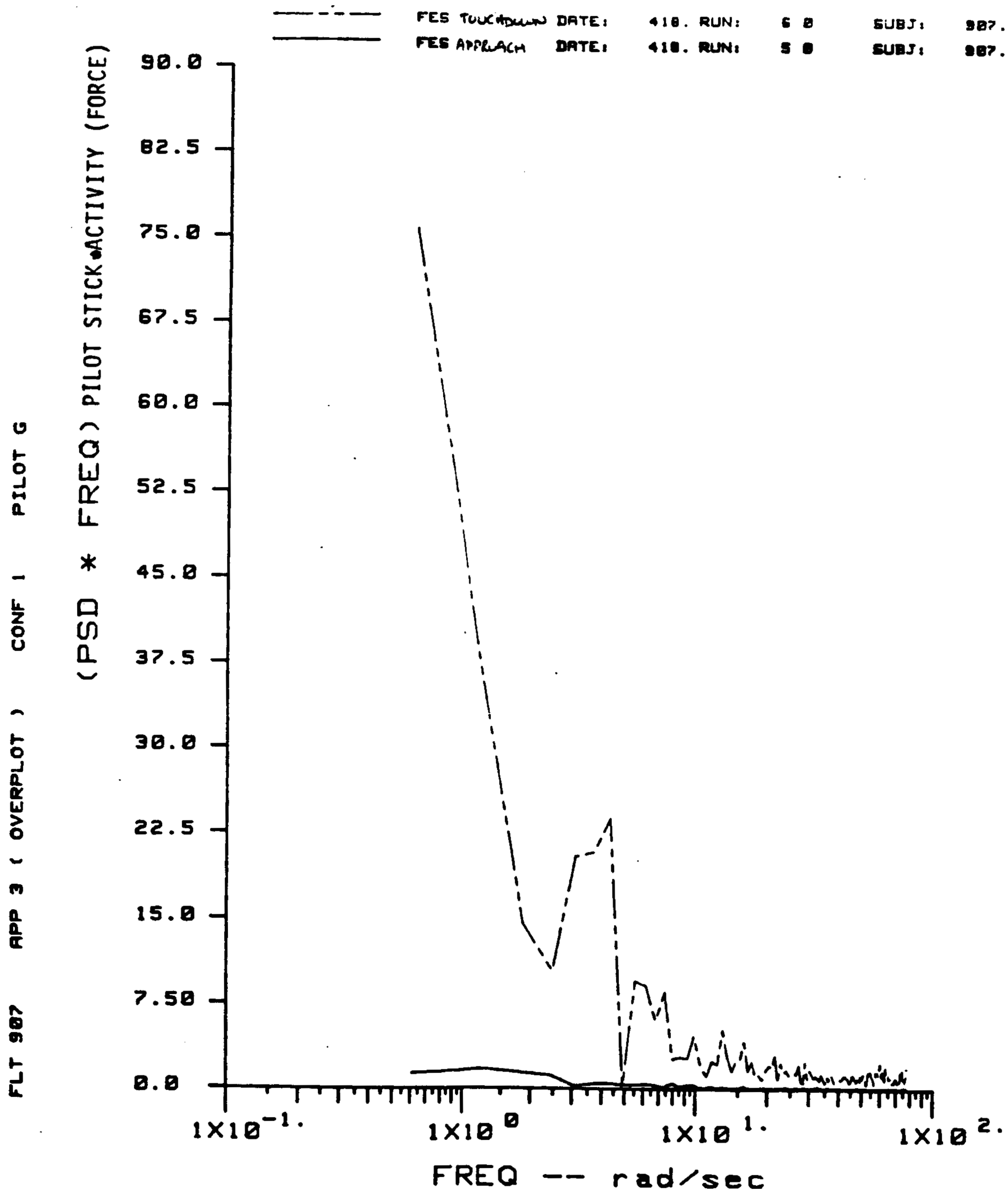


Figure 33. PSD, CONFIGURATION 1, PILOT G

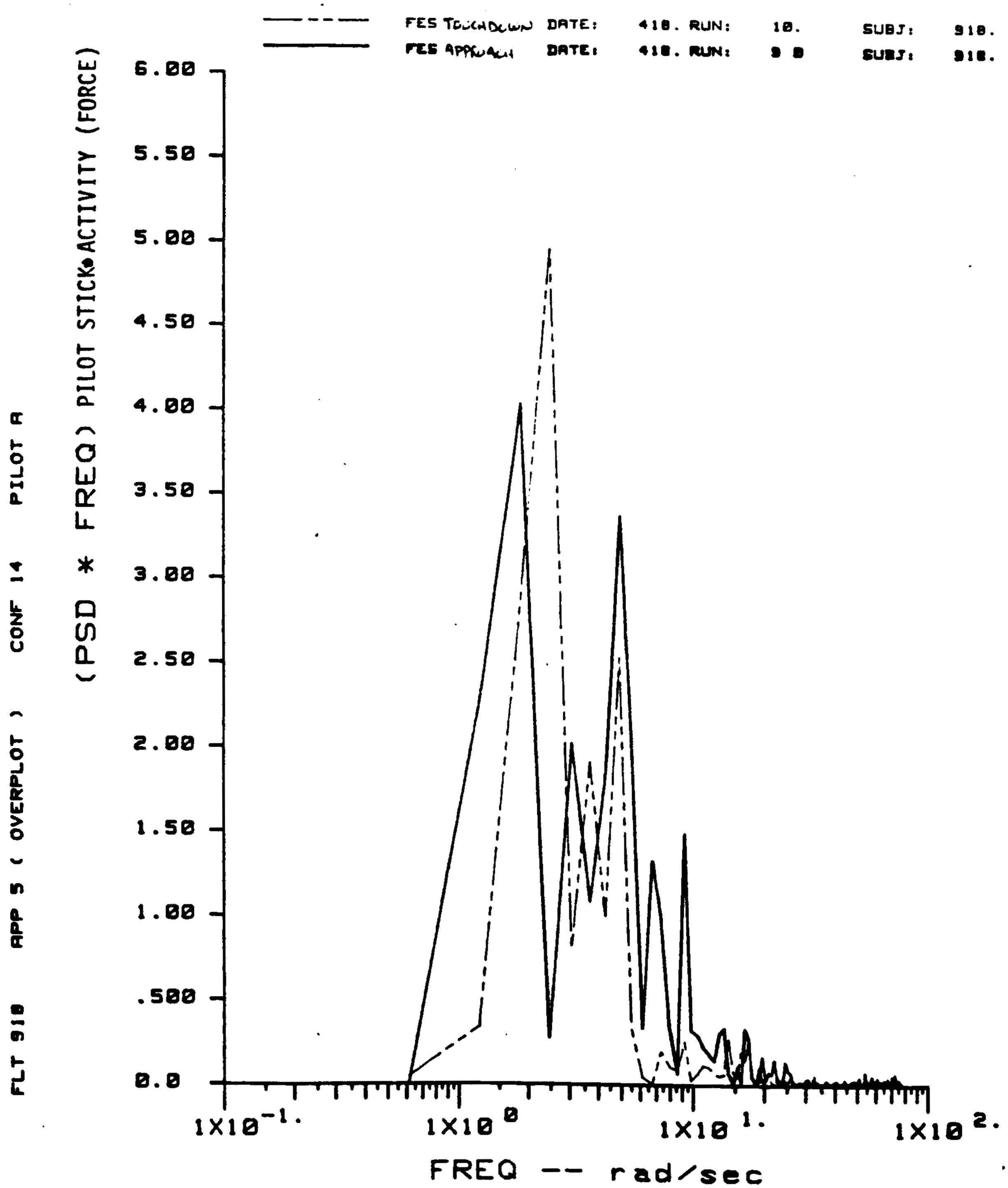


Figure 34. PSD, CONFIGURATION 14, PILOT A

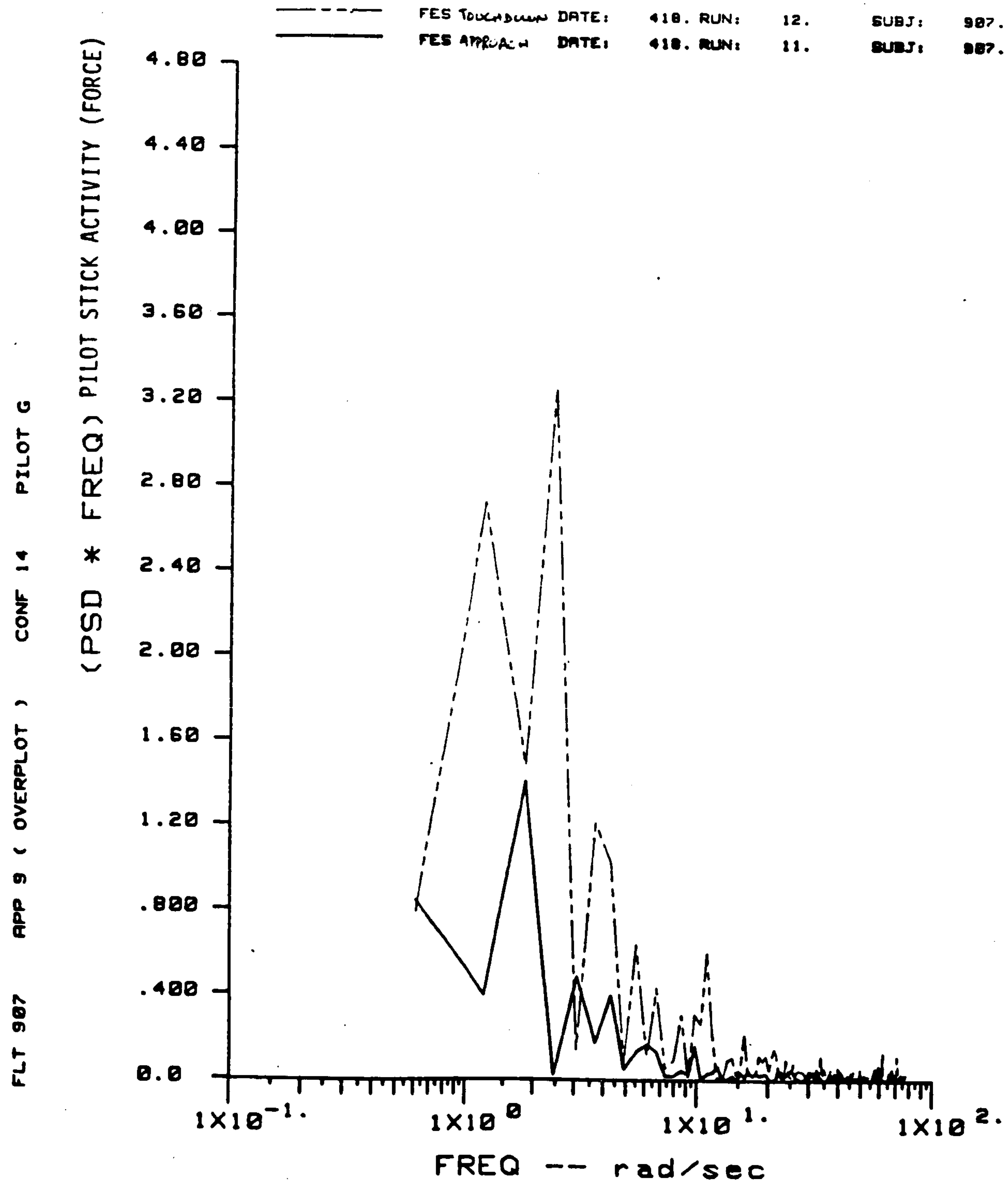


Figure 35. PSD, CONFIGURATION 14, PILOT G

5.4 CONCLUSIONS AND RECOMMENDATIONS

Conclusions

1. The result of the experiment strongly suggest that the pilot wants the pitch attitude to be useable in determining the flight path. The difference between attitude and flight path angle, or angle of attack, appears to be the most important variable associated with the flying qualities of an airplane. Because the pilot generally does not sense angle of attack, the angle of attack response should be "well behaved".

2. The flying qualities specification, MIL-F-8785(C) specifies short period frequency requirements of the angle of attack response of the vehicle as a function of n/α and defines what is meant by a "well behaved" angle of attack response in the short term.

3. Results suggest that pilots prefer a pitch rate command system only if the angle of attack response is well behaved as defined by the ω_n vs n/α requirements of MIL-F-8785(C). Because the angle of attack response of a pitch rate command system is dominated by $1/\tau_{\theta 2}$ in the short term, pilot preference for a pitch rate command system is a function of $1/\tau_{\theta 2}$.

4. In the long term, a primary requirement appears to be that pitch angle and flight path angle changes be in harmony, i.e.:

   A relative high frequency phugoid appears not to be objectionable if $\dot{\alpha}(t) \doteq 0$ in the long term, i.e., if $\Delta\theta = \Delta\gamma$.

   Pilot preference is for pitch rate command/attitude "hold" only if precise attitude "hold" also results in precise flight path angle "hold".

5. Results of the flight test program suggest that the Category C precision requirements, as they appear in MIL-F-8785(C), are adequate for approach. Higher precision, perhaps as defined by the Category A requirements, appear to be required for the flare and landing segment of the task.

6. The results suggest that a wide range of piloting technique can be accommodated by:

    1. An angle of atack command configuration showing large phugoid mode residues in θ and γ.

    2. A pitch rate command configuration showing very little phugoid mode residues in α and γ.

Recommendations

1. Validation of the lower ω_n vs n/α CAP boundary is critical to flying qualities and flight control system design. The vehicle for this experiment should be configured both as an angle of attack and pitch rate command system with no phugoid mode residue in either angle of attack or pitch rate in order to separate phugoid and short period modal effects.

2. The effect of phugoid mode residue in $\alpha(t)$ for both angle of attack and pitch rate command systems is an important element in the establishment of design of criteria requirements. Indications are that a phugoid mode effect will be substantial. A flight experiment to define allowable phugoid mode residues is highly recommended.

3. The values of $1/\tau_{\theta_2}$ as a function of ω_{sp} to establish the boundary between angle of attack and rate command preference should be verified.

4. Additional command configurations directed toward the optimum use of multiple controllers is critical to future flight control system design.

5. A strong flight control system design trend is toward the higher order rate command, attitude hold system. Proper design criteria for this type of system should be established.

    a) Direct lower order (4th) and higher order (5th) comparisons should be made.

    b) Phugoid mode residue limits and $1/\tau_{\theta_2}$ requirements should be established.

6. A similar study of the $\dot{\gamma}(\gamma)$ command system configuration should be undertaken. Because the $\dot{\gamma}$ system seems to be gaining in popularity, the requirements for excellence should be established.

7. A flight control system design trend is toward decoupled velocity or "auto throttle" behavior. Because of this trend, an investigation of decoupled $\gamma/\Delta V$ systems should be undertaken.

8. Associated with recommendation (7) is the requirement for flying qualities or dynamic behavior that would best allow a pilot to fly a wind shear profile. What FCS configurations satisfy the shear/flying qualities requirements?

9. An angle of attack command system is an $\alpha(t)$ regulator and could be expected to reduce the vehicle angle of attack response sensitivity in turbulence. A rate command system would reduce attitude excursions in turbulence. An important flying qualities area of investigation involves the design of the command/response configuration in the presence of turbulence and wind shears.

Section 6
PREDICTIVE CRITERIA RESULTS AND ANALYSES

6.1 TIME DOMAIN RESULTS AND ANALYSES

6.1.1 Introduction

In the past, time delays have been considered, for the most part, autonomous contributors to flying qualities. Attempts were made to specify various levels of time delay as being acceptable or not acceptable for a given flight task. This has resulted in some confusion in that examples keep croping up that seem to be exceptions to the rule; i.e., large aircraft in general and, more recently, the X-29. In both of these cases, Level 1 performance was achieved with time delays that were excessive according to most of the literature.

Experience with the Calspan Learjet has indicated that the effect of a given amount of time delay is dependent on a number of parameters; i.e., open-loop frequency and damping, command gain, and feel system dynamics. A hypothesis was developed that the common denominator was the sensitivity of the aircraft to pilot inputs. This program provided an opportunity to examine these time delay effects in a formal manner.

The results show that as pitch sensitivity is increased, tolerance to time delay decreases. In fact, by a proper choice of lower pitch sensitivity, Level 1 performance could be maintained in the flared landing task with time delays from 150 ms to delays in excess of 300 ms. With higher sensitivity, configuration with Level 1 performance at 150 ms degraded to Level 2 at 200 ms.

The time delay data from this program has allowed development of parameters that provide the designer with a method of predicting time delay effects using simple measurements of computer generated time histories. These time delay and sensitivity metrics have been applied to the previously developed time domain predictive criteria and the result is a more mature criteria which, when applied to seven flying qualities programs, successfully predicted 104 of 129 configurations by flying qualities Level (81%). The flying qualities of 77 (60%) of these configurations were predicted to within ±1 unit of the average Cooper-Harper rating.

6.1.1.1 Data Base Description

Data Base Composition

The original intent of this program was to refine the time domain predictive criteria of the previous pitch rate program (Reference 5) by expanding the data base of that program and to include time delay effects. The first part of this effort was to be a search of existing data bases and develop a flight program test matrix that would fill the gaps of existing data rather than duplicating previous data.

The search revealed that much of the existing data base was not applicable, and further that very little usable data was practically available outside programs previously conducted by Calspan. In order to be usable for the time domain predictive criteria refinement:

- Pilot ratings must be available for the flared landing task.
- The pilot ratings must apply to the longitudinal dynamics.
- Time histories must be available, or practical to reconstruct, for angle of attack, pitch rate, pilot station normal acceleration, pitch attitude, flight path, controller input, etc.
- The number of variables in the flight configuration must be limited so that cause and effect may be determined.

In many cases, the pilot assigned an overall rating for approach and landing and the flared landing task rating was not available. In other cases although the program was directed at longitudinal dynamics, the task was up-and-away, not landing. Until recently, programs did not utilize angle of attack time histories in the analysis and they were not practically available. Other previous programs did not limit the number of variables (one tries to get as much out of each flight test configuration as possible) or allowed the pilot freedom in selecting sensitivity. Consequently, although most previous data can be useful in testing a predictive criteria, very little previous data was found to be useful in developing one.

During the data search it became apparent that time delay data must be generated by this program. It was also determined that creating data for criteria refinement alone was not totally efficient. The time delay test matrix could be accomplished with a minimum of nine (9) and a maximum of eighteen (18) configurations. There were other research efforts that could achieve data from this program and the configurations from these other programs could also be valuable for criteria development outside the time delay area. Consequently, a flight control program was conducted in parallel with the criteria development effort and the results of both programs were used for criteria development and refinement.

The test matrix was consequently composed of nineteen (19) flight control configurations and thirteen (13) time delay configurations for a total of thirty two (32) configurations.

Data Base Discussion

The details of the time delay and flight control test matrix are discussed in Section 2. The coverage of this matrix was excellent in that only three (3) of the thirty-two (32) configurations did not receive multiple evaluations. The average coverage was three evaluations, conducted by three different pilots, per configuration. This allowed a much more complete normalization of the data than is possible in most flying qualities programs and enhanced trends which led to more valid analysis.

Without the increased coverage provided by additional sponsors during the flight phase of the program, the analysis would have been only partly successful.

6.1.2 Time Domain Results

The results of this program, as in most flying qualities research programs, was pilot rating data and comments. On-board digital recording served primarily to assure that the model was correct and model following was achieved. The pilot comments were recorded on video and audio tape as well as being manually recorded by the flight test engineer. Comments were also monitored by the safety pilots and any inconsistencies with comments versus ratings were discussed on the spot. This provided faster learning curves for evaluation pilots who were not highly experienced with the rating scale.

6.1.2.1 Pilot Ratings

Individual pilot ratings are shown in Figure 36 and 37. Figure 36 depicts the ratings for the flight control configurations and Figure 37 depicts ratings for the time delay/sensitivity configurations. (In Figure 37 the configurations are presented out of numerical order to better present trends as sensitivity is increased at each level of time delay.)

A brief study of these figures will demonstrate quite vividly the importance of multiple evaluations by multiple pilots in flying qualities research. This is shown particularly well in Figure 37, where one knows intuitively (and from previous flight test) that performance should decrease with increases in time delay and sensitivity from the norm. Pilot B follows the expected trend quite closely. Pilot A does for the most part but in the case of Configuration 28, he rates it an HQR of 3 in one evaluation and an HQR of 8 in another (both evaluations under similar atmospheric conditions). In the case of Pilot C, configurations off nominal sensitivity pay a large penalty and in some cases overshadow time delay effects. Pilot F in Configurations 25, 22, and 28 shows a revese trend by preferring higher sensitivities as time delay is increased.

When one is faced with analyzing data that resulted from single evaluations this sort of pilot scatter can lead to non-results. One is then faced with an agonizing study of pilot comments in an attempt to identify causes of pilot variance, (there are many which are quite valid) and even if the causes are found the result is still a rather poor base for quantitative analysis. On the other hand average values from a larger pilot population and repeat evaluations can result in very meaningful data. One can manually fair in averages on Figure 37 and see definite trends. Figure 38 is a plot of the average pilot ratings of Figure 37 and clearly shows valid trends and additionally quantifies the results.

An additional value of repeat evaluations and multiple pilots is the ability to recognize anomalous data. An example of this can be seen in Figure 36. Of the 14 evaluations conducted by Pilot G, seven were at wide variance with other evaluations. There are a number of explanations for the variance, however, the usefulness of this type data to criteria development is

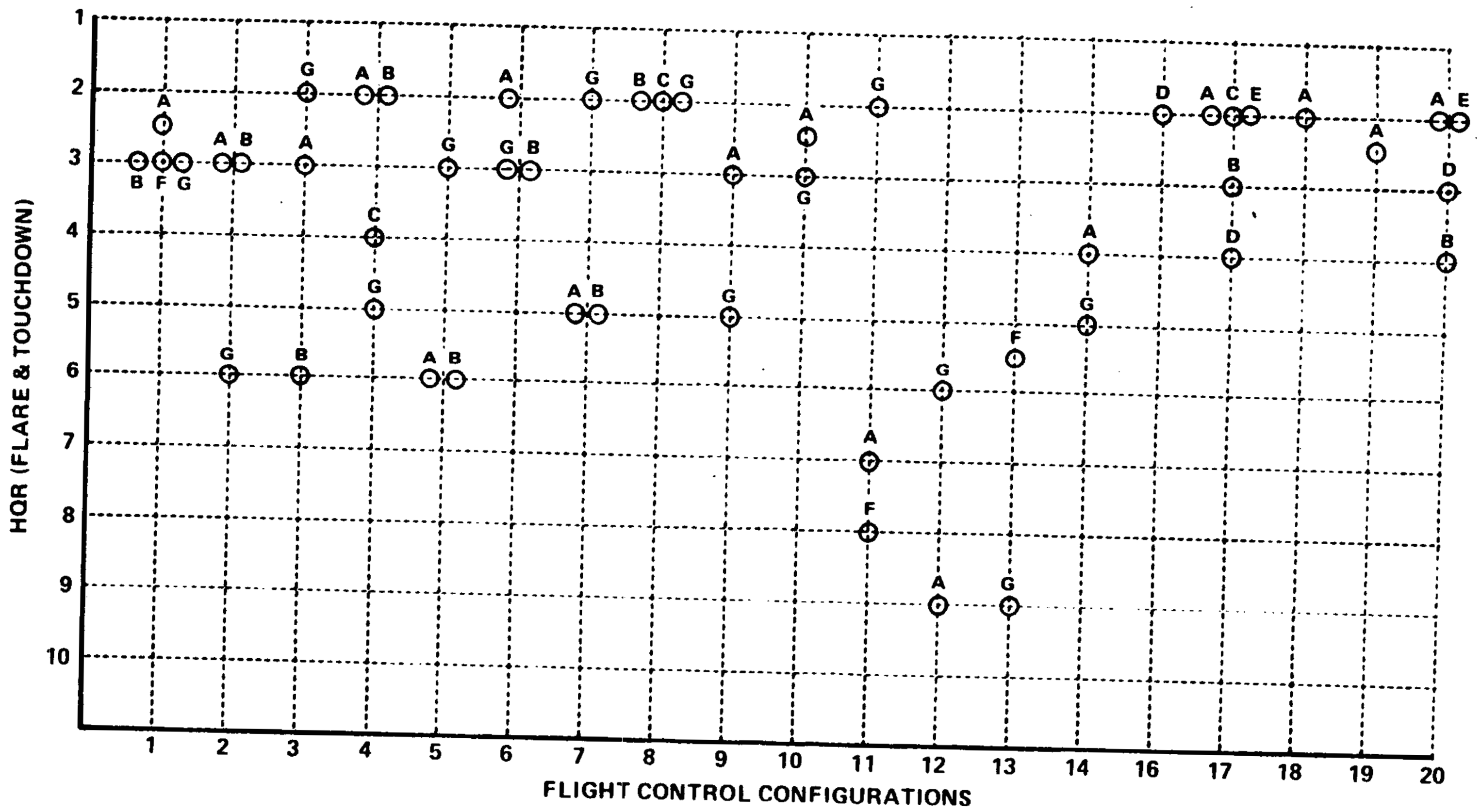


Figure 36 INDIVIDUAL PILOT RATINGS, FLARE AND TOUCHDOWN (FLIGHT CONTROL CONFIGURATIONS)

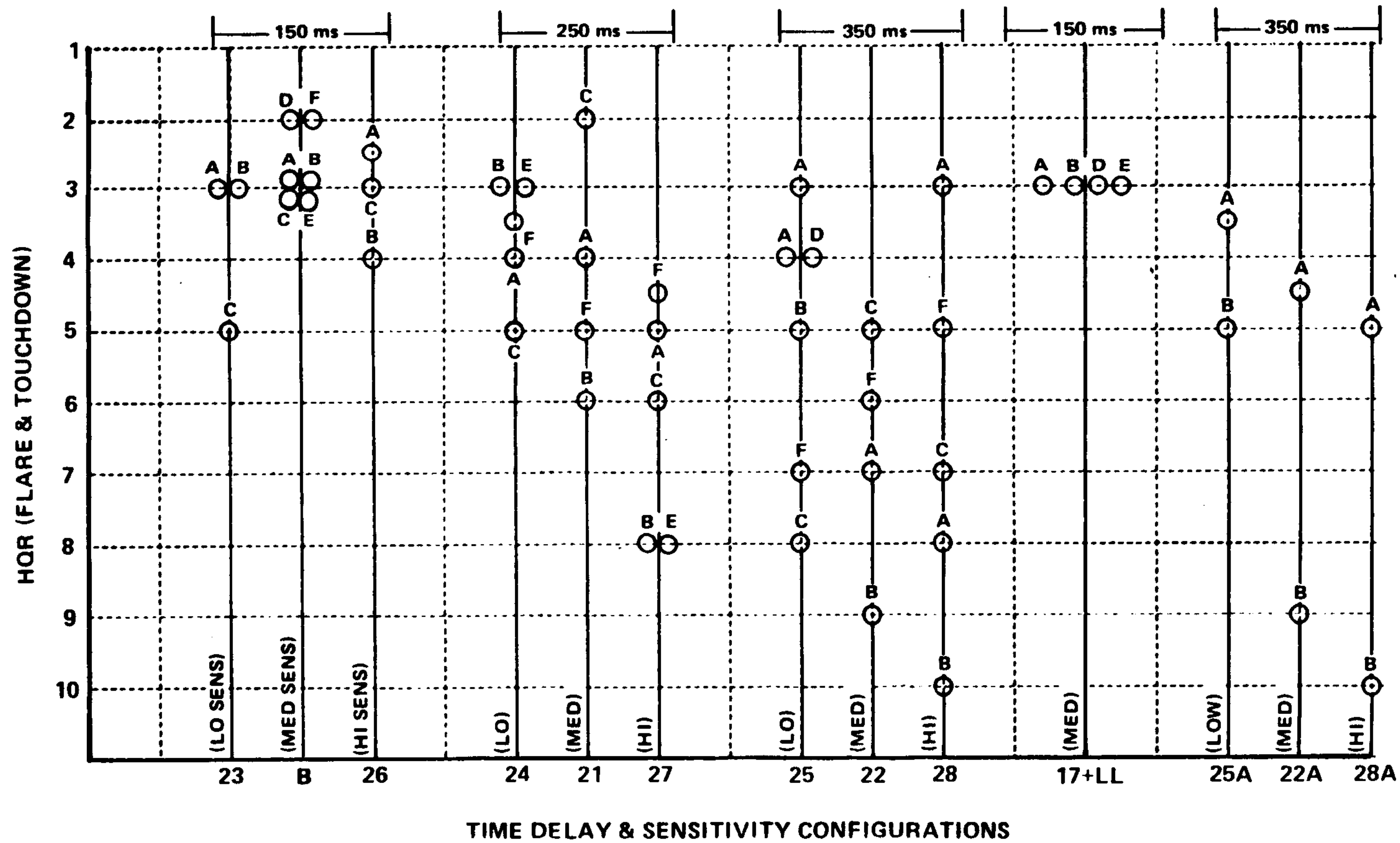


Figure 37 INDIVIDUAL PILOT RATINGS, FLARE AND TOUCHDOWN (TIME DELAY CONFIGURATIONS

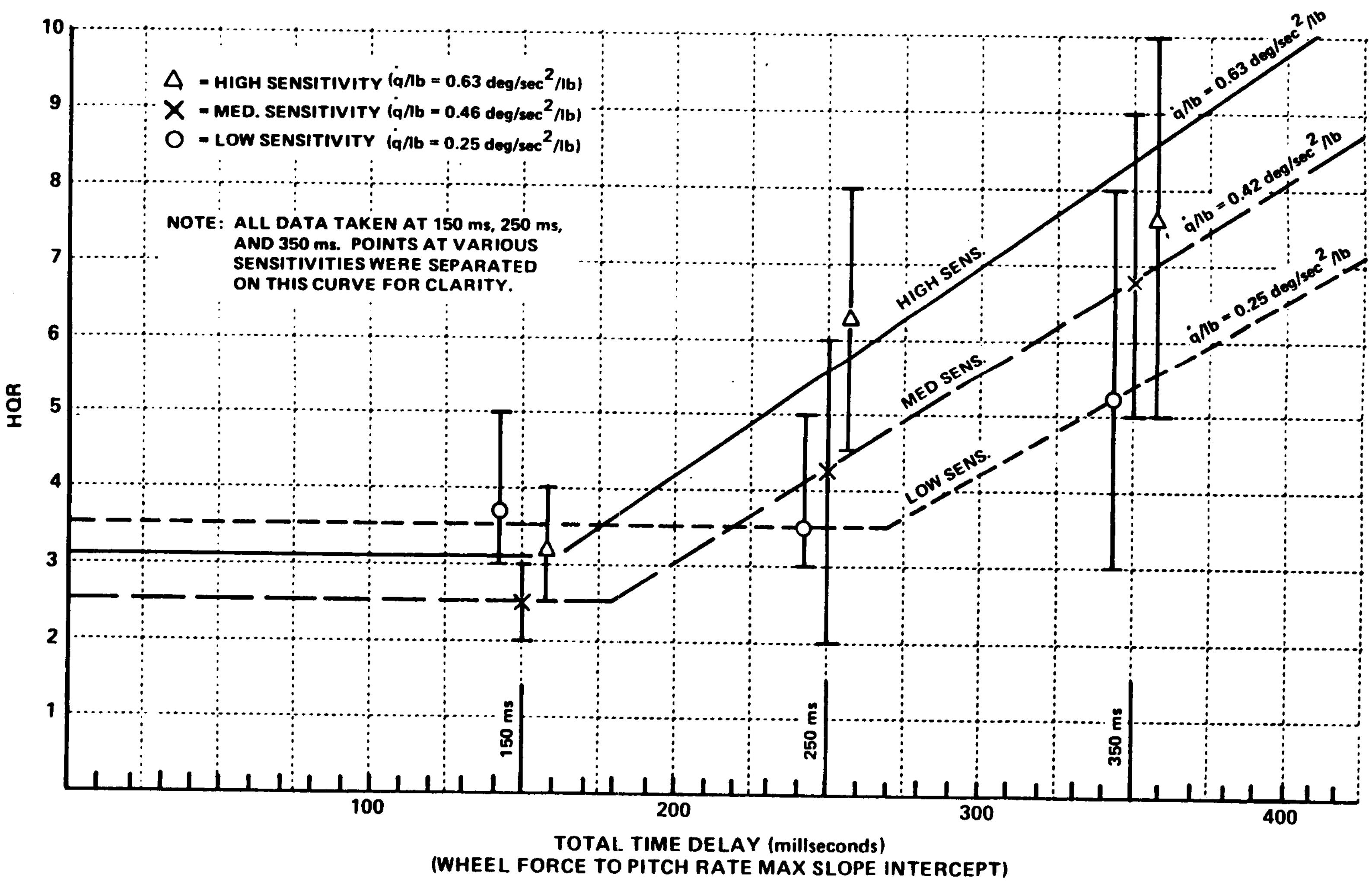


Figure 38 EFFECTS OF TIME DELAY AND PITCH SENSITIVITY FOR WHEEL CONTROLLERS

questionable. In a data base of this size one has the luxury of recognizing the data as anomalous and discard it for the purpose of criteria quantification. Another case can be seen with pilot A and Configuration 28. In one instance an HQR of 3 was assigned and in another instance an HQR of 8. The explanation in this case is rather straight forward. If the pilot does not make inputs or makes only open-loop type inputs time delays will not affect performance. In some cases pilots can get by with this technique. On the other hand, when the same pilot is forced into the loop for some reason; i.e., turbulence, slightly higher level of aggressiveness, etc., time delays can greatly affect performance. It would be unconservative to develop time delay criteria based on data resulting from open-loop pilot techniques.

In any event, all data is shown for the purposes of other types of analysis or to allow the reader to take issue with the writer's selection of anomalous data points.

6.1.2.2 Discussion of Results

The primary results for the purposes of this analysis were those obtained from the time delay matrix. These results were required to quantify time delay and sensitivity effects. The results of the flight control matrix were needed to test the refined predictive criteria.

The pilot ratings of the time delay matrix, Figure 37, were averaged and plotted versus time delay at various levels of pitch sensitivity in Figure 38. Pilot rating scatter is also depicted. As expected, the data showed strong correlation between pitch sensitivity, $\dot{q}$/lb (deg/sec^2/lb), and time delay (max slope intercept of pitch rate response to column force input). A study of Figure 38 shows that at the minimum time delay (150 ms) slight penalties were paid for sensitivities lower and higher than nominal, however, all three sensitivities resulted in Level 1, or borderline Level 1, performance. At the high sensitivity any increase in time delay resulted in a rapid decrease in flying qualities performance (higher HQR), and Level 1 performance could be maintained only out to approximately 180 ms. At the normal sensitivity, Level 1 performance was maintained out to approximately 220 ms. At the low sensitivity Level 1 performance could be maintained out to 270 ms and three of the six pilots who evaluated the low sensitivity/high time delay configuration achieved Level 1 and borderline Level 1/Level 2 performance at 350 ms.

It appears there are trade-offs available to the designer between time delay and pitch sensitivity for the flared landing task.

6.1.3 Time Domain Analysis

The first goal of the analysis was to develop time delay and pitch sensitivity metrics that would quantify the effects and have applicability to the time domain predictive criteria. The second goal was to refine the criteria using these metrics. The final goal of the analysis was to apply the refined criteria to a data base acquired from a number of previous programs in order to test its viability.

6.1.3.1 Time Delay and Sensitivity Metric Development

Study of Figure 38 provided some insight into the parameters that should be addressed in the desired metrics:

- Slope of sensitivity lines of Figure 38.

  The slope of the nominal sensitivity line is 0.025 HQR/ms. This was compared to data from References 1, 3, 4, 5, and 6. The flying qualities degradation with time delay was remarkably consistent and matched favorably with the value of 0.025 HQR/ms from Figure 38. A final value of 0.02 HQR/ms was chosen as a median value for use in the metric.

- Maximum acceptable time delay; i.e., time delay threshold.

  Experience with various in-flight simulators and using these simulators to train test pilot students to identify and quantify time delays and effects led the writer to initially choose a value of around 100 ms as an acceptable threshold for the flared landing task, consequently, 100 ms was chosen as a metric value.

- Sensitivity effect in time delay.

  Figure 38 shows a degradation, at a given time delay, in flying qualities with sensitivities greater than 0.42 $\dot{q}$/lb (deg/sec²/lb) and delayed degradation at sensitivities less than 0.42 $\dot{q}$/lb. Consequently, it was determined that a metric "modifier" of $\frac{\dot{q}/lb}{0.45}$ should be used (0.45 being a nominal value).

As a result of the above, the time delay metric developed was:

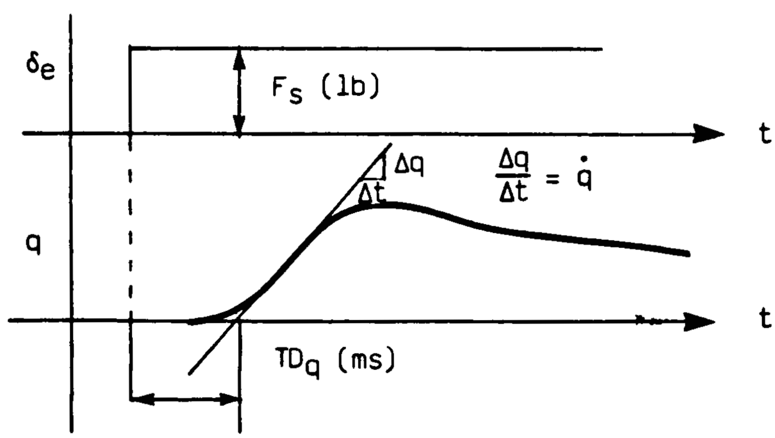
$$TD' = 0.02\,(TD_q - 100)\,\frac{\dot{q}/lb}{0.45}$$

Where: TD' = time delay metric (Δ HQR).

TD_q = time from wheel controller pitch force step input to maximum slope intercept of the resul tant pitch rate response (ms).

$\dot{q}$/lb = pitch sensitivity measured by the maximum slope of the pitch rate responses to a step input divided by the pitch force (deg/sec²/lb).

and when TD_q is less than 100 ms, let $TD' = 0$



Consideration was given to other time delay measurements; i.e., max slope intercept of the angle of attack response, gamma response, N_{z_p} response, etc. The use of angle of attack time delay proved promising, however, the results when applied to the data base were no better than those of pitch rate time delay, and as pitch rate time delay is an established parameter in the literature it was chosen as the metric.

A sensitivity metric was required due to the fact that off optimum sensitivity alone can degrade flying qualities. Figure 38 showed that an optimum sensitivity for wheel controllers was close to 0.42 deg/sec²/lb. Also from Figure 38, near Level 1 performance was attainable when sensitivity was limited to ±0.2 deg/sec²/lb. Consequently, the sensitivity metric developed was:

$$\dot{q}' = \left| \frac{\dot{q}/\mathrm{lb} - 0.45}{0.2} \right|$$

where:

$\dot{q}'$ = sensitivity metric (Δ HQR).

$\dot{q}$/lb = pitch sensitivity (deg/sec²/lb as previously defined).

0.45 = nominal optimum sensitivity

0.2 = an allowable sensitivity range

Analysis of the data base of References 1, 3, 4, 5, and 6 disclosed that time delay effects could be predicted reasonably well for wheel controllers. This is because there is a larger data base for wheel controllers and this particular program used wheel controllers to specifically address time delay and sensitivity. Programs using other controllers; i.e., Reference 1, LAHOS, using a center stick and Reference 3, NLR, using a side stick, varied many parameters including sensitivity. In order to properly identify sensitivity effects specifically, data must be used from a program structured to identify sensitivity effects, specifically.

An attempt was made to identify sensitivity levels for a center stick controller (Reference 1, LAHOS). In the LAHOS program the pilots were allowed to choose sensitivity while other powerful variables were also being introduced. Consequently, it was very difficult to separate sensitivity effects alone.

A first approximation for center stick controllers resulted:

$$TD' = 0.02\,(TD_{\dot q} - 100)\ \frac{\dot q/lb}{2.8}$$

$$\dot q' = \left| \frac{\dot q/lb - 2.8}{1.2} \right|$$

In the first approximation it appeared that there is a 6 to 1 increase in sensitivity required for center stick controllers versus wheel controllers.

Subsequently in the analysis, application of these center stick controller metrics with the predictive criteria to the LAHOS data will show fair predictions of pilot ratings, however, the results were the worst of the data sets tested. This is felt to be due to the lack of adequate sensitivity data for center stick controllers, and to the large number of single evaluations of the configurations.

6.1.3.2 Predictive Criteria Refinement

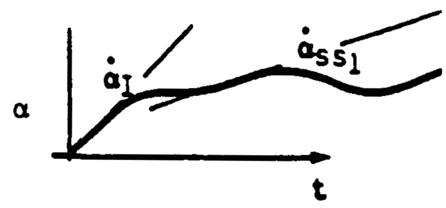
A review of the original time domain predictive criteria (Reference 5) would show:

$$PHQR = 1.7\,\dot\alpha' - 1.44\,N_{z_p}' + 0.55\,T_{\dot\alpha} + 3.9$$

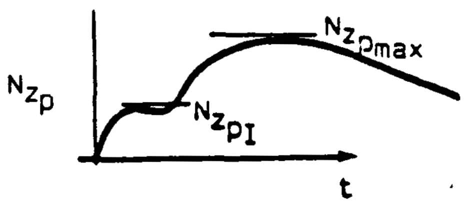
where:

PHQR = Predicted Handling Qualities Rating

$\dot\alpha' = \dot\alpha_{ss}/\dot\alpha_I$ where:

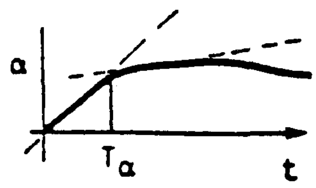


$N_{z_p} = N_{z_{p_I}}/N_{z_{p_{max}}}$ where:



$$T'_{\alpha} = |T_{\alpha} - 1| \left[\frac{|\dot{\alpha}'| + 0.05}{|N'_{z_p}| + 0.05} \right]$$

and



when: $T'_{\alpha} > 6$ let $T'_{\alpha} = 6$, and T_{α} is defined by

and when: $T_{\alpha} < 1$ let $T'_{\alpha} = 0$

(The 0.05 terms prevent unrealistic contributions of T_{α} when $\dot{\alpha}$ or N_{z_p} are very small or zero. The $(T_\alpha - 1)$ term indicates that $T_\alpha = 1$ sec is a near optimum value for flared landing task.)

3.9 = A bias term.

In the pitch rate program of Reference 5 pitch sensitivity was held essentially constant at 0.45 deg/sec^2/lb and time delay was 170 ms except for a few configurations. Consequently, the criteria was not valid for configurations of other sensitivities and time delays. The 3.9 bias term included the effects of 170 ms time delay and near optimum sensitivity.

The first step of the refinement was to simply add the new metrics and modify the bias term. A "criteria perfect" flight control system should yield an HQR of 2 (similar to Configuration 2-1 of the LAHOS program, Reference 1). In the "criteria perfect" airplane the angle of attack response is flat at the steady state; i.e., $\dot{\alpha}' = 0$, the pilot is located near the center of rotation; i.e., $N'_{z_p} = 0$, T_α is one second or less; i.e., $T'_\alpha = 0$, the time delay is 100 ms or less; i.e., $TD' = 0$, and if the pitch sensitivity is nominal, $\dot{q}' = 0$. Consequently:

$$PHQR = 0 - 0 + 0 + 0 + 0 + \text{Bias} = 2$$

Therefore, $\text{Bias} = 2$

The refined time domain criteria then becomes:

$$PHQR' = 1.7\,\dot{\alpha}' - 1.44\,N'_{z_p} + 0.55\,T'_{\alpha} + TD' + \dot{q}' + 2.0$$

(With the terms as previously described and TD' and $\dot{q}'$ modified as required to accomodate center stick or wheel controllers)

The refined criteria was then applied to a number of data bases from this and previous programs (results presented in a subsequent section). The intent was to refine the parameters of the criteria as necessary. As it turned out the criteria as presented provided predictions well below pilot scatter levels and no further modifications were made.

6.1.3.3 Application and Limitations of the Time Domain Predictive Criteria

The criteria was originally designed to apply to flight control designs that fall within or close to Level 1 boundaries of the MIL-F-8785 $\omega_{n_{sp}}$ and N_z/α requirements (the logic was that a flight control designer can normally locate the poles and zeros where he pleases, consequently, this criteria was not meant to handle low damping ratios or excessively high or low frequencies. The problem was that the MIL Spec was originally designed to handle conventional pole/zero locations, i.e., conventional short period and phugoid roots, and newer designs (pitch rate command, etc.) were meeting MIL Spec. requirements as far as short period pitch rate pole locations are concerned, but having problems with flared landing performance. The first criteria, that resulting from Reference 5, identified the problems associated with non-conventional pole/zero locations. It also could identify effects of pilot location. An unplanned asset was its ability to identify low short period frequency problems by virtue of the T_{α}' term. It would not accomodate low damping ratios, time delays, pitch sensitivity variations, or excessively high short period frequencies.

The revised criteria can accomodate time delay and pitch sensitivity effects. A by-product of the sensitivity parameter is that excessively high short period frequencies will pay a penalty by virtue of resultant higher sensitivities, however, this has not been adequately tested.

There are four areas of known limitations to the criteria two of which have been previously discussed:

1. Lightly damped configurations are not applicable by design.

2. Divergent configurations are not applicable by design.

3. "Decoupled" configurations are not applicable. During this program Configuration 5 through 10 were decoupled in nature. That is they were angle of attack command and pitch rate command (by virtue of the flat steady state response of these parameters). This was accomplished by use of an additional controller, in this case direct lift flaps, which provide increased lift with out changing the angle of attack. In this program, lift could be changed at constant speed and constant angle of attack by the direct lift flaps. Use of the criteria for decoupled configurations will require further modification. Decoupled configurations were shown on the curves but were not used for testing the criteria in subsequent sections of the analysis.

4. Pitch controllers other than wheel and center stick are not yet applicable due to the lack of sufficient sensitivity data. In fact, the sensitivity parameters for center stick are not yet sufficiently refined. The NLR data base, Reference 3, could not be used in testing the criteria due to lack of sensitivity data for the controller used in that experiment.

It should be re-emphasized that the criteria is applicable to the flared landing task. It was not designed to accomodate up-and-away tasks such as cruise or air-to-ground tracking, etc. It has only limited application to the approach task and has never been tested for the non-flared landing task.

6.1.4 Time Domain Predictive Criteria Testing

The criteria was tested using flight data from seven programs (including the current landing program). Configurations from these programs that were not appropriate to the criteria were not tested; i.e., divergent or lightly damped configurations, decoupled configurations, controllers other than wheel or center stick, and task other than flared landing.

6.1.4.1 Landing Program Results

The criteria was applied to the data of the present landing program in the form:

$$PHQR' = 1.7\ \dot{\alpha}' - 1.44\ N'_{z_p} + 0.55\ T'_{\dot{\alpha}} + TD' + \dot{q}' + 2.0$$

where α', N'_{z_p} and $T'_{\dot{\alpha}}$ are as previously defined, and TD' and $\dot{q}'$ are adjusted for wheel controllers, i.e.,

$$TD' = 0.02\ (TD_q - 100)\ \frac{\dot{q}/lb}{0.45}$$

$$\text{and} \quad \dot{q}' = \left| \frac{\dot{q}/lb - 0.45}{0.2} \right|$$

The calculations for the predicted pilot ratings and the actual pilot ratings are shown in Table 14. The results are plotted on Figures 39 and 40, Figure 39 reflecting only the time delay matrix data and Figure 40 showing the total program results. The curves are a plot of average HQR from the flight data (AHQR) versus predicted HQR (PHQR').

Figure 39, the time delay matrix (13 configurations), shows the application of the criteria to Level 1 flight control systems, both angle of attack command and pitch rate command. The results are summarized by:

CONFIGURATIONS PREDICTED TO WITHIN 1 HQR	CONFIGURATIONS PREDICTED TO WITHIN 2 HQR	CONFIGURATIONS PREDICTED BY FLYING QUALITIES LEVEL OR WITHIN 1 HQR
11 of 13 or 85%	13 of 13 or 100%	13 of 13 or 100%

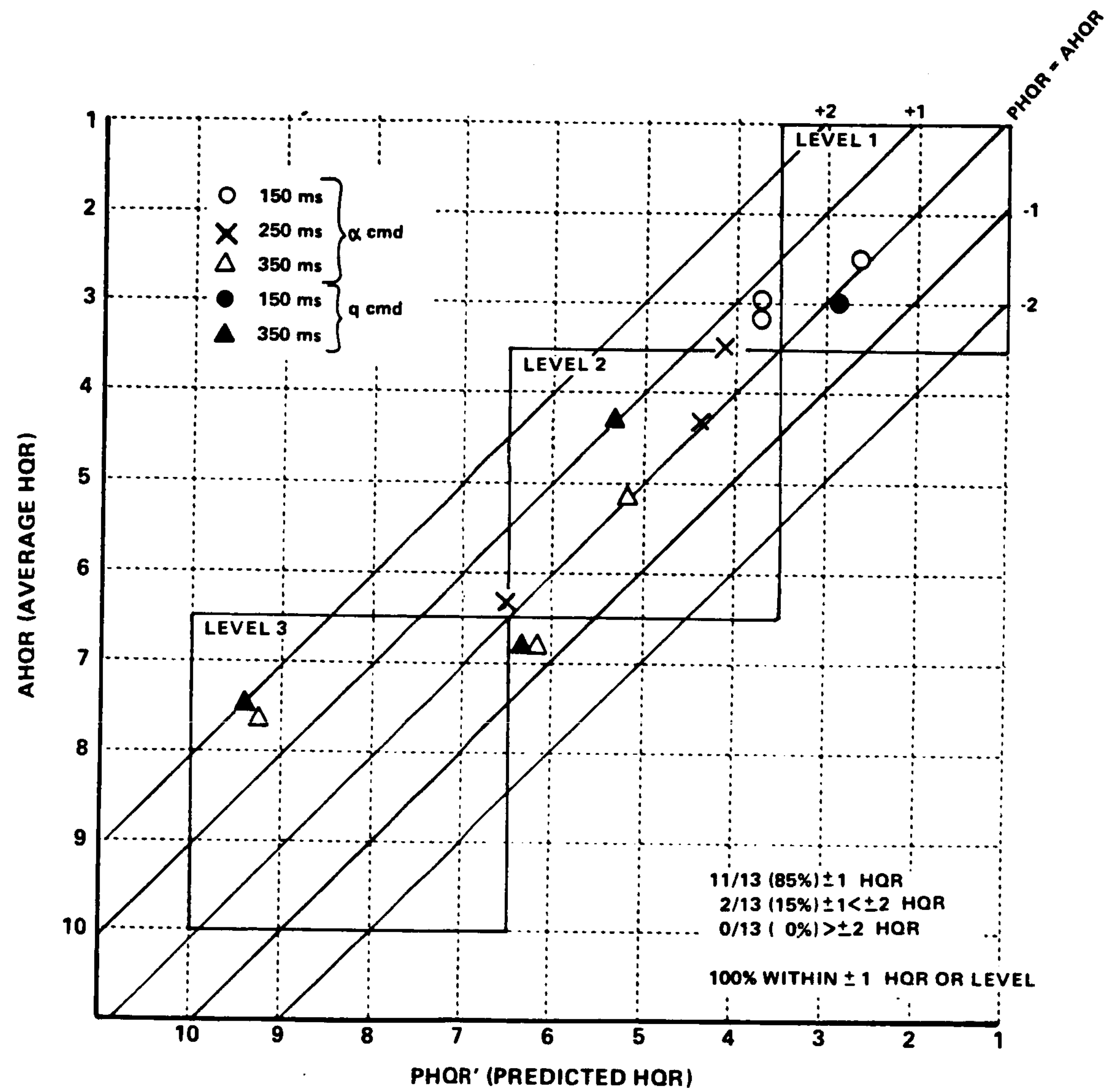


Figure 39 REFINED TIME DOMAIN CRITERIA APPLIED TO THE LANDING PROGRAM (TIME DELAY CONFIGURATIONS)

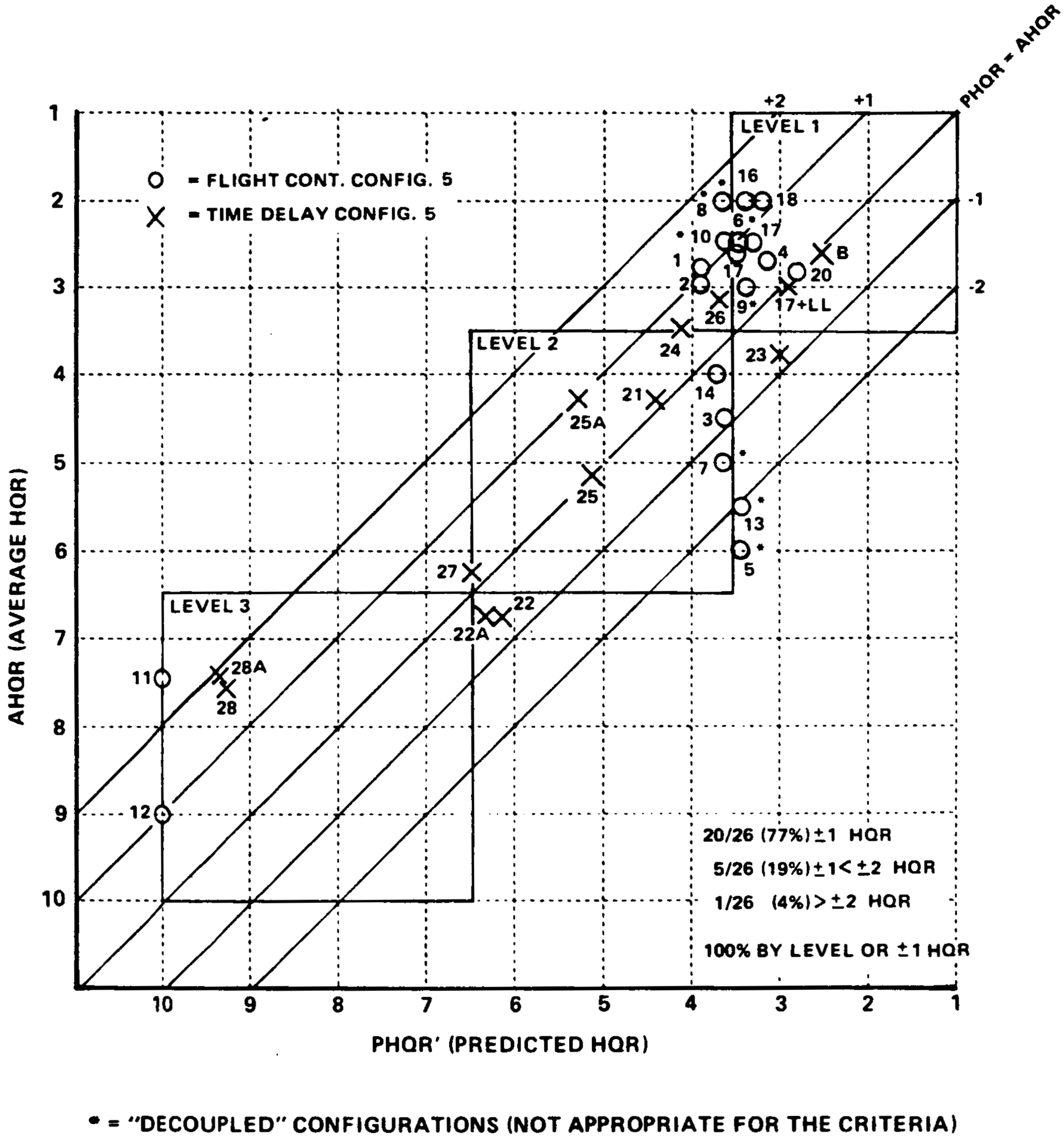


Figure 40 REFINED TIME DOMAIN CRITERIA APPLIED TO THE LANDING PROGRAM (ALL CONFIGURATIONS)

Table 14
LANDING PROGRAM - TIME DOMAIN CRITERIA APPLICATION

$(PHQR' = 1.7\ \dot{\alpha}' - 1.44\ N'_{z_p} + 0.55\ T'_{\alpha} + TD' + \dot{q}' + 2.0)$

(where TD' and $\dot{q}'$ used wheel controller sensitivity factors)

CONFIG	$1.72\ \dot{\alpha}'$	$1.44\ N'_{z_p}$	$0.55\ T'_{\alpha}$	$\dot{q}$/lb	TD_q	TD'	$\dot{q}'$	PHQR'	AHQR	HQR	$1/\tau_{\theta_2}$
1	0	0	0.11	0.5	170	1.8	0.25	3.9	2.8	(2.5)(3)	0.5
2	0.49	0	0	0.5	150	1.4	0.25	3.9	3.0	(3)(3)	0.5
3	0.17	0	0	0.5	150	1.4	0.25	3.6	4.5	(3)(6)	0.5
4	0.19	0	0	0.5	150	1.1	0.25	3.3	2.7	(2)(2)(4)	0.5
5	0	0	0	0.5	150	1.4	0.25	3.4	6.0	(6)(6)	0.9
6	0	0	0	0.5	150	1.4	0.25	3.4	2.0	(2)(2)	0.9
7	0	0	0	0.5	160	1.6	0.25	3.6	5.0	(5)(5)	0.9
8	0.07	0	0.10	0.5	150	1.4	0.25	3.6	2.0	(2)(2)	0.9
9	0	0	0	0.5	150	1.4	0.25	3.4	3.0	(3)	0.5
10	0.15	0	0	0.5	150	1.4	0.25	3.6	2.5	(2.5)	0.5
11	1.0	0	3.3	0.5	250	3.6	0.25	10.0	7.5	(7)(8)	0.5
12	0.85	0	3.3	0.5	300	4.7	0.25	10.0	9.0	(9)	0.5
13	0	0	0	0.5	150	1.4	0.25	3.4	5.5	(5.5)	2.0
14	0.34	0	0	0.5	150	1.4	0.25	3.7	4.0	(4)	2.0
16	0	0	0	0.5	150	1.4	0.25	3.4	2.0	(2)	1.0
17	0.29	0.43	0.10	0.5	150	1.4	0.25	3.4	2.6	(2)(3)(2)(4)(2)	0.72
18	0.19	0.43	0.10	0.5	150	1.4	0.25	3.3	2.0	(2)	0.72
19	0.10	0.58	0	0.5	170	1.8	0.25	3.3	2.5	(2.5)	0.72
20	0	0.58	0	0.5	150	1.4	0.25	2.8	2.8	(2)(4)(3)(2)	0.72
23	0	0.58	0	0.25	150	1.6	1.0	3.0	3.7	(3)(3)(5)	0.75
B	0	0.58	0	0.42	150	1.1	0.15	2.5	2.6	(3)(3)(3)(2)(3)(2)(2)	0.75
26	0	0.58	0	0.63	150	2.3	0.9	3.7	3.2	(2.5)(4)(3)	0.75
24	0	0.58	0	0.25	250	2.7	1.0	4.1	3.5	(4)(3)(5)(3)(3.5)	0.75
21	0	0.58	0	0.42	250	3.0	0.15	4.4	4.3	(4)(6)(2)(5)	0.75
27	0	0.58	0	0.63	250	5.1	0.9	6.5	6.3	(5)(8)(6)(8)(4.5)	0.75
25	0	0.58	0	0.25	350	3.8	1.0	5.2	5.2	(3)(4)(5)(8)(4)(7)	0.75
22	0	0.58	0	0.42	350	4.8	0.15	6.2	6.8	(7)(9)(5)(6)	0.75
28	0	0.58	0	0.63	350	7.9	0.9	9.3	7.6	(8)(10)(7)(8)(5)	0.75
17+LL	0.14	0.72	0	0.5	150	1.1	0.25	2.8	3.0	(3)(3)(3)(3)	0.72
25A	0.14	0.72	0.04	0.25	350	2.8	1.0	5.3	4.3	(3.5)(5)	0.72
22A	0.14	0.72	0.04	0.42	350	4.7	0.15	6.3	6.8	(4.5)(9)	0.72
28A	0.14	0.72	0.04	0.63	350	7.0	0.9	9.4	7.5	(5)(10)	0.72

Figure 40, all configurations applicable to the criteria (26 configurations), shows the application of the criteria to many different flight control designs and effective time delays. The results are summarized by:

CONFIGURATIONS PREDICTED TO WITHIN 1 HQR	CONFIGURATIONS PREDICTED TO WITHIN 2 HQR	CONFIGURATIONS PREDICTIONS THAT MISSED BY MORE THAN 2 HQR	CONFIGURATIONS PREDICTED BY FLYING QUALITIES LEVEL OR WITHIN 1 HQR
20 of 26 or 77%	25 of 26 or 96%	1 of 26 or 4%	26 of 26 or 100%

6.1.4.2 Pitch Rate Program Results

The refined criteria was applied to all configurations of the Pitch Rate Program (Reference 5), again using wheel controller sensitivity factors.

Calculations are shown in Table 15 and Figure 41 shows the application of the refined criteria to the original data base. The results are summarized by:

CONFIGURATIONS PREDICTED TO WITHIN 1 HQR	CONFIGURATIONS PREDICTED TO WITHIN 2 HQR	CONFIGURATIONS PREDICTIONS THAT MISSED BY MORE THAN 2 HQR	CONFIGURATIONS PREDICTED BY FLYING QUALITIES LEVEL OR WITHIN 1 HQR
17 of 27 or 63% (Previous Criteria 14 of 27 or 52%)	24 of 27 or 89% (Previous Criteria 23 of 27 or 85%)	3 of 27 or 11% (Previous Criteria 4 of 27 or 15%)	21 of 27 or 78% (Previous Criteria 19 of 27 or 70%)

Although most of the pitch rate configurations were flown at the same sensitivity and time delay there were a few at greater time delays and off nominal sensitivity. Consequently, the refined criteria improved the prediction results.

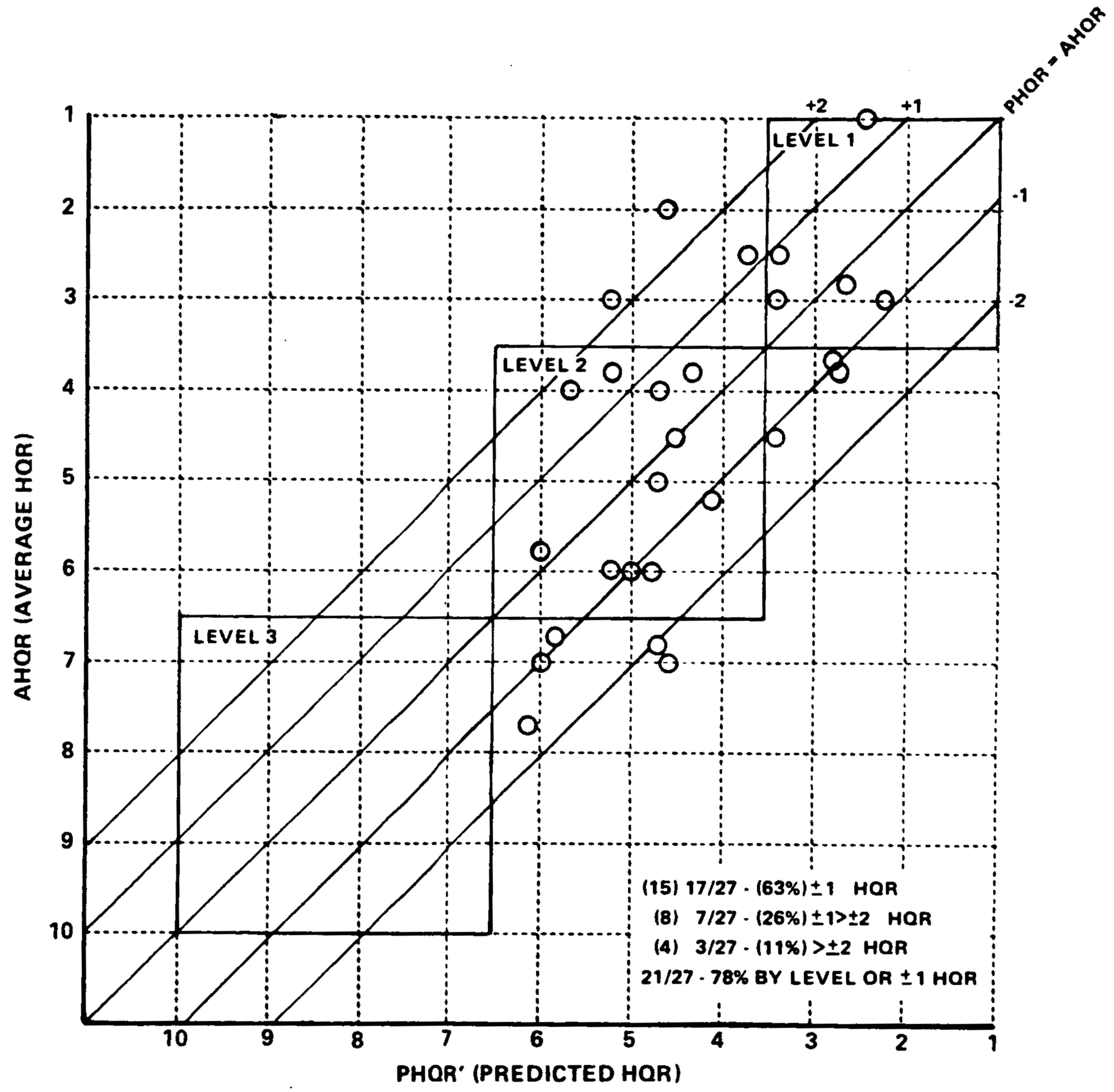


Figure 41 REFINED TIME DOMAIN CRITERIA APPLIED TO THE PITCH RATE PROGRAM

Table 15
LAHOS PROGRAM - TIME DOMAIN CRITERIA APPLICATION

$$(PHQR' = 1.7\,\dot{\alpha}' - 1.44\,N'_{z_p} + 0.55\,T'_{\alpha} + TD' + \dot{q}' + 2.0)$$

(where TD' and $\dot{q}'$ used wheel controller sensitivity factors)

CONFIG	PHQR (From Pitch Rate Program)	Δ PHQR (Corr. for New Bias Term)	TD_q	q/lb	TD'_q	$\dot{q}'$	PHQR'	AHQR
1-1-1	4.5	2.6	170	0.58	1.8	0.7	5.0	6.0
1-2-2	4.1	2.2	170	0.58	1.8	0.7	4.7	6.8
1-3-7	3.9	2.0	170	0.58	1.8	0.7	4.5	4.5
2-1-1	5.4	3.5	170	0.34	1.1	0.6	5.2	6.0
2-2-2	4.5	2.6	170	0.34	1.1	0.6	4.3	3.8
3-1-3	6.2	4.3	170	0.32	1.0	0.7	6.0	5.8
3-2-4	5.4	3.5	170	0.32	1.0	0.7	5.2	3.8
4-1-1	2.6	0.7	170	0.52	1.6	0.4	2.7	3.8
4-2-2	3.3	1.4	170	0.52	1.6	0.4	3.4	2.5
4-3-7	3.5	1.6	170	0.8	2.5	1.8	5.9	7.0
4-3-1-1	3.3	1.4	170	0.8	2.5	1.8	5.7	4.0
5-1-1	3.6	1.7	170	0.32	1.0	0.7	3.4	4.5
5-2-2	3.9	2.0	170	0.32	1.0	0.7	3.7	2.5
6-1-1	4.9	3.0	170	0.32	1.0	0.7	4.7	5.0
6-1-1-1	3.6	1.7	170	0.32	1.0	0.7	3.4	3.0
6-2-1	3.1	1.2	170	0.45	1.4	0	2.8	3.7
6-2-1-1	2.7	0.8	170	0.45	1.4	0	2.2	3.0
7-1-4	3.0	1.1	170	0.40	1.2	0.3	2.6	2.8
8-1-5	4.3	2.4	170	0.30	0.9	0.8	4.1	5.2
8-1-5-1	4.8	2.9	170	0.30	0.9	0.8	4.6	2.0
8-2-5	6.1	4.2	170	0.17	0.5	1.4	6.1	7.7
8-2-5-1	4.6	2.7	170	0.17	0.5	1.4	4.6	7.0
8-3-5	5.4	3.5	220	0.17	0.9	1.4	5.8	6.7
8-3-5-1	4.8	2.9	220	0.17	0.9	1.4	5.2	3.0
8-4-6	2.6	0.7	170	0.33	1.0	0.7	2.4	1.0
8-5-5	4.7	2.4	210	0.30	1.5	0.8	4.7	6.0
8-5-5-1	4.3	2.4	210	0.30	1.5	0.8	4.7	4.0

6.1.4.3 LAHOS Program Results

The LAHOS data (Reference 1) was the most difficult to utilize due primarily to the lack of sensitivity data for center stick controllers. In LAHOS the pilots were given freedom to choose pitch sensitivity. At the same time there were variations in other major parameters. Optimum sensitivity was estimated by fairing straight lines through data that contained many other variables. The nominal sensitivity chosen was 2.8 $\dot{q}$/lb (deg/sec²/lb) as opposed to 0.45 $\dot{q}$/lb for wheel controllers, or a ratio of approximately 6 to 1. Sensitivity variation was set at 1.2 versus 0.2 for wheel controllers.

Consequently, the criteria applied to the LAHOS data was:

$$PHQR' = 1.7\,\dot{\alpha}' - 1.44\,N_{z_p}' + 0.55\,T_{\alpha}' + TD' + \dot{q}' + 2.0$$

where: α', N_{z_p}' and T_{α}' are as previously defined

and:

$$TD' = 0.02\,(TD_q - 100)\,\frac{\dot{q}/lb}{2.8}$$

$$\dot{q}' = \left|\frac{\dot{q}/lb - 2.8}{1.2}\right|$$

Pilot trends were very difficult to analyze from the LAHOS data. The author feels that this is due in large part to the size of the test matrix. Twenty one (21), of the thirty two (32) configurations that were tested for the criteria were single evaluations; i.e., one pilot, one evaluation. If the reader will review Figures 36 and 37 it will be apparent why single evaluations are quite risky for research purposes. If one were limited to a single random choice of the pilot ratings from each evaluation of Figures 36 and 37, it becomes obvious why some flying qualities programs reach indefinite conclusions. One could have obtained by random choice, results that show that as time delay was increased for a given configuration flying qualities get better! In fact, this trend was found in some of the LAHOS data.

In any event, all the LAHOS data that was appropriate to the criteria was used and no effort was made to attempt to identify anomalous data. The results were plotted on Figure 42 with the supporting data in Table 16. The results are summarized by:

CONFIGURATIONS PREDICTED TO WITHIN 1 HQR	CONFIGURATIONS PREDICTED TO WITHIN 2 HQR	CONFIGURATIONS PREDICTIONS THAT MISSED BY MORE THAN 2 HQR	CONFIGURATIONS PREDICTED BY FLYING QUALITIES LEVEL OR WITHIN 1 HQR
13 of 32 or 41%	23 of 32 or 72%	9 of 32 or 28%	24 of 32 or 75%

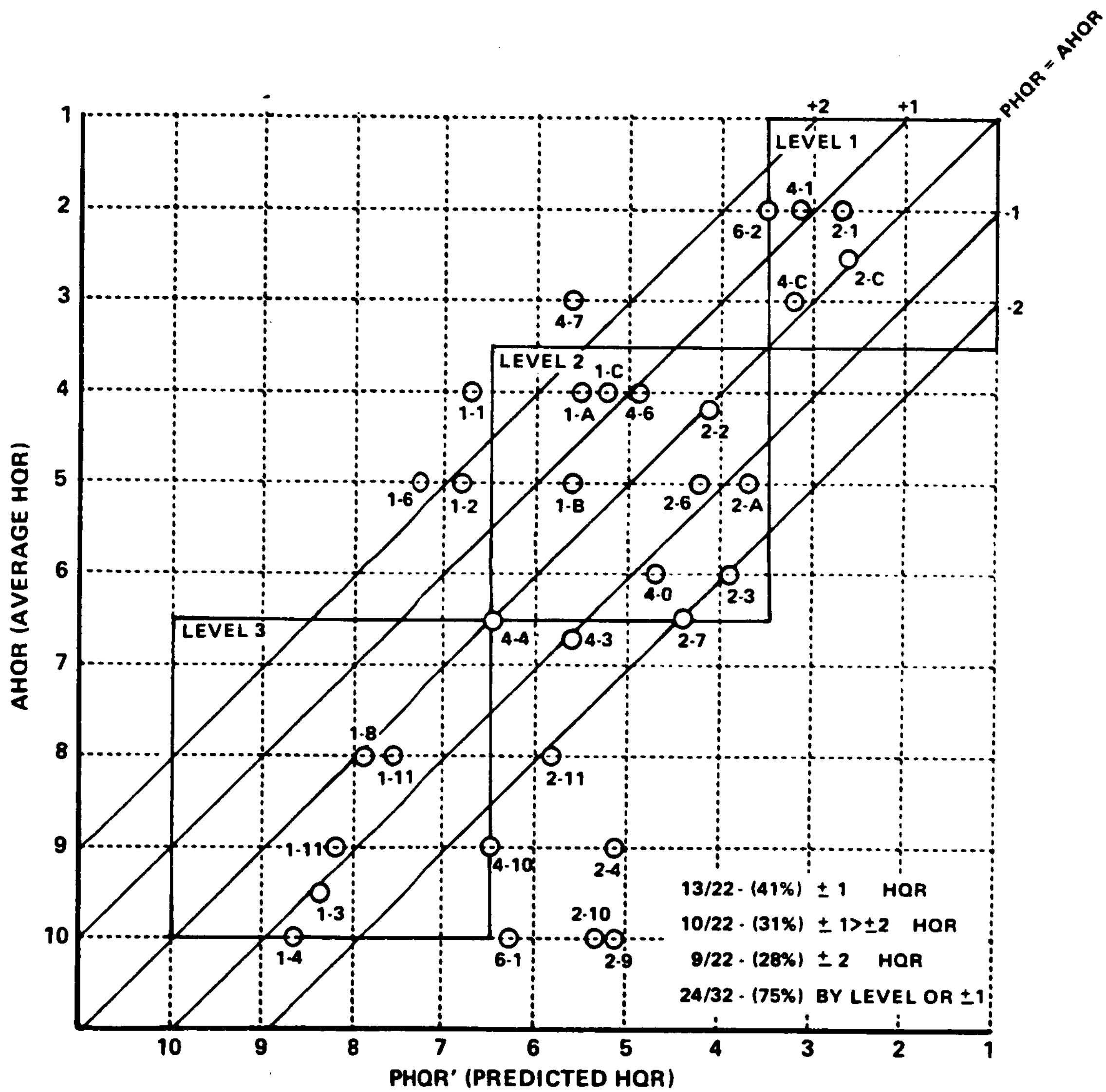


Figure 42 REFINED TIME DOMAIN CRITERIA APPLIED TO THE LAHOS PROGRAM

Table 16
LAHOS PROGRAM - TIME DOMAIN CRITERIA APPLICATION

$$(PHQR' = 1.7\,\dot{\alpha}' - 1.44\,N'_{z_p} + 0.55\,T'_{\alpha} + TD' + \dot{q}' + 2.0)$$

$$\text{and:}\quad TD' = (0.02)\left|TD_q - 100\right|\frac{\dot{q}/1b}{2.8},\quad \dot{q}' = \left|\frac{\dot{q}/1b - 2.8}{1.2}\right|$$

CONFIG	$1.72\,\dot{\alpha}'$	$1.44\,N'_{z_p}$	$0.55\,T'_{\alpha}$	$\dot{q}/1b$	TD_q	TD'	$\dot{q}'$	PHQR'	AHQR	HQR	SAFETY PILOT
1-A	0.3	0	3.0	3.0	100	0	0.2	5.5	4	(4)	3
1-B	0.3	0	3.0	2.5	100	0	0.25	5.6	5	(5)	4
1-C	0.3	0	3.0	2.8	100	0	0	5.3	4	(4)(4)	2,3
1-1	0.3	0	3.3	1.5	100	0	1.1	6.7	4	(4)(4)	3,4
1-2	0.3	0	2.8	1.6	150	0.6	1.1	6.8	5	(5)	6
1-3	0.2	0	3.3	0.9	300	1.3	1.6	8.4	9.5	(9)(10)	8,8
1-4	0.5	0	3.3	1.0	300	1.4	1.5	8.7	10	(10)	10
1-6	0.4	0	3.3	1.8	160	0.8	0.8	7.3	5	(5)	6
1-8	0.3	0	3.3	0.8	200	0.6	1.7	7.9	8	(8)	9
1-11	0.3	0	3.3	0.9	250	1.0	1.6	8.2	9	(9)	8
2-A	0	0	0	4.8	100	0	1.7	3.7	5	(4)(6)	3,5
2-C	0	0	0	3.5	100	0	0.6	2.6	2.5	(4)(1.5)(1.5)(1.3)	3,2,2,2
2-1	0	0	0.3	2.0	100	0	0.7	2.7	2	(2)(2)	1,2
2-2	0	0	0	1.6	200	1.1	1.0	4.1	4.2	(4)(4.5)	2,2
2-3	0	0	0.2	1.6	160	0.7	1.0	3.9	6	(6)	4
2-4	0	0	0.4	1.5	250	1.6	1.1	5.1	9	(9)	8
2-6	0	0	0.10	2.2	200	1.6	0.5	4.2	5	(5)	4
2-7	0	0	0.1	1.8	220	1.5	0.8	4.4	6.5	(7)(6)	6,6
2-9	0	0	0.2	1.0	300	1.4	1.5	5.1	10	(10)	8
2-10	0	0	0.3	0.8	320	1.3	1.7	5.3	10	(10)	10
2-11	0	0	0.1	2.1	300	3.0	0.7	5.8	8	(8)	8
4-C	0.1	0	0.3	3.8	100	0	0.8	3.2	3	(3)(3)	4,2
4-0	0.2	0	0.7	5.0	100	0	1.8	4.7	6	(6)	4
4-1	0.1	0	0.8	3.0	100	0	0.2	3.1	2	(2)	3
4-3	0.1	0	1.6	1.6	180	0.9	1.0	5.6	6.7	(5)(8)(7)	5,3,7
4-4	0.1	0	2.0	1.0	210	0.8	1.5	6.4	6.5	(7)(6)	7,5
4-6	0.1	0	0.8	2.6	200	1.9	0.2	5.0	4	(4)	3
4-7	0.1	0	0.8	1.5	250	1.6	1.1	5.6	3	(3)	3
4-10	0.1	0	1.6	0.6	320	0.9	1.8	6.4	9	(9)	9
4-11	0.1	0	1.2	2.6	320	4.1	0.2	7.6	8	(8)	7
6-1	0.1	0	1.0	0.9	350	1.6	1.6	6.3	10	(10)	
6-2	0.1	0	0.6	1.8	100	0	0.8	3.5	2	(2)	

6.1.4.4 Large Aircraft Program Results

The Large Aircraft program (Reference 4) was used as a source of data for the original time domain criteria of the Pitch Rate program. In the original application these configurations with time delays in excess of 170 ms were excluded as they exceeded the scope of the original criteria. Also, when used to test the original criteria, no criteria parameter addressed sensitivity.

In this case the refined criteria (modified for wheel controller) was applied to thirty four (34) configurations, and included time delays from 170 ms to 410 ms. Those configurations excluded were those that were divergent and those where the pilot ratings were based on the approach and not the flared landing task.

The results are plotted on Figure 43, with supporting data in Table 17. The results are summarized by:

CONFIGURATIONS PREDICTED TO WITHIN 1 HQR	CONFIGURATIONS PREDICTED TO WITHIN 2 HQR	CONFIGURATIONS PREDICTIONS THAT MISSED BY MORE THAN 2 HQR	CONFIGURATIONS PREDICTED BY FLYING QUALITIES LEVEL OR WITHIN 1 HQR
18 of 34 or 53% (Previous Criteria 5 of 16 or 31%)	31 of 34 or 91% (Previous Criteria 13 of 16 or 81%)	3 of 34 or 9% (Previous Criteria 3 of 27 or 19%)	24 of 34 or 71% (Previous Criteria 10 of 27 or 63%)

These results are considered to be a significant improvement over the original criteria, particularly considering the much more broad scope of application.

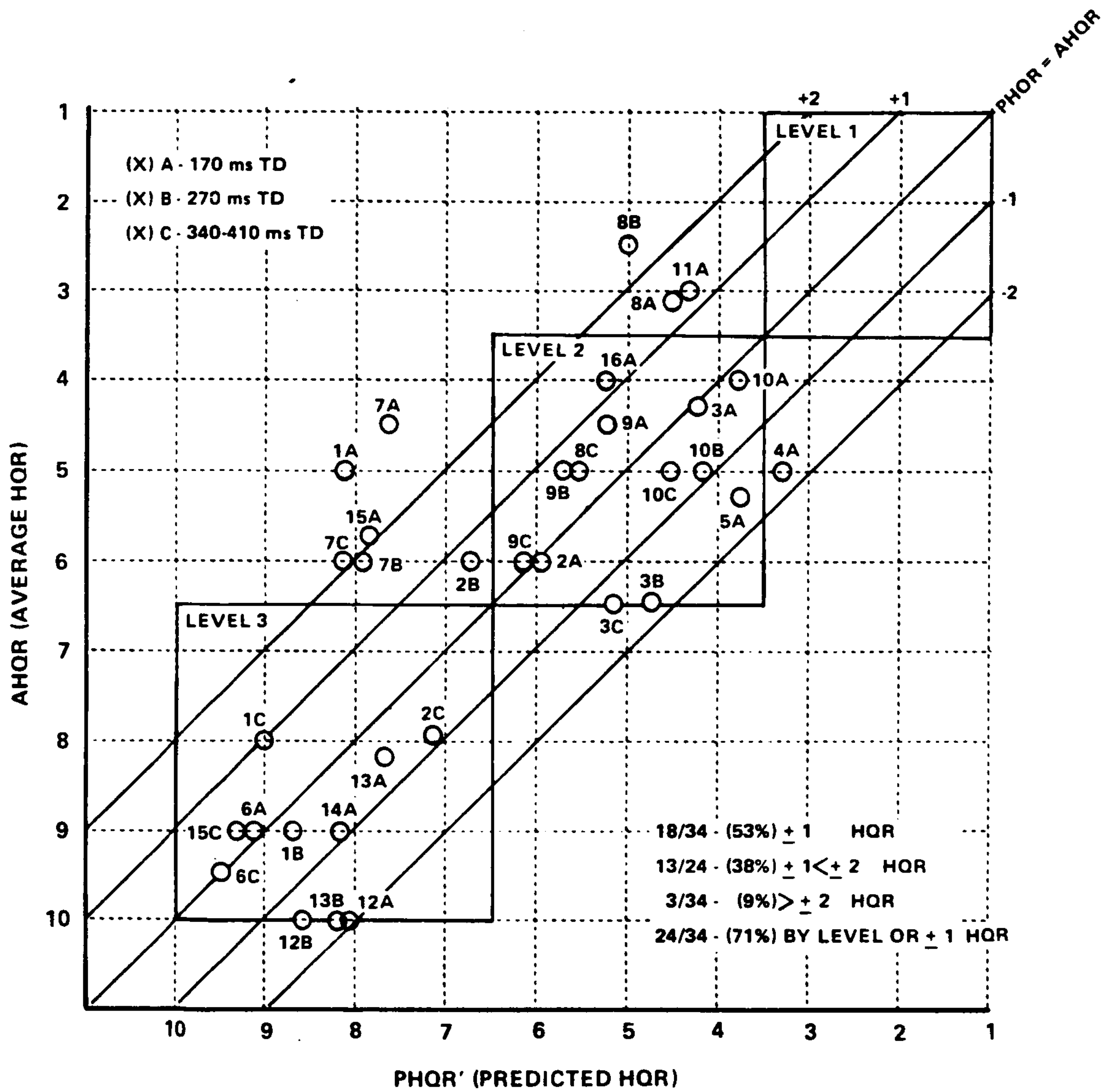


Figure 43 REFINED TIME DOMAIN CRITERIA APPLIED TO THE LARGE AIRCRAFT PROGRAM

Table 17
LARGE AIRCRAFT PROGRAM - TIME DOMAIN CRITERIA APPLICATION

$$(PHQR' = 1.7\,\dot{\alpha}' - 1.44\,N'_{z_p} + 0.55\,T'_{\alpha} + TD' + \dot{q}' + 2.0)$$

(where TD' and $\dot{q}'$ used sensitivity factors adjusted to wheel controllers)

CONFIGURATION	1.72 $\dot{\alpha}'$	1.44 N'_{z_p}	0.55 T'_{α}	$\dot{q}$/1b	TD_q	TD'	$\dot{q}'$	PHQR'	AHQR	HQR
LAT										
1A (lo α) A	1.14	0.35	3.3	0.11	170	0.3	1.7	8.1	5.0	(5)
1B (lo α) B	1.14	0.35	3.3	0.11	270	0.8	1.7	8.6	9.0	(9)
1C (lo α) C	1.14	0.35	3.3	0.11	340	1.2	1.7	9.0	8.0	(9)(7)
2A (med α) A	0.7	0.43	1.8	0.11	170	0.3	1.7	6.1	6.0	(6)
2B (med α) B	0.7	0.43	1.8	0.11	270	0.8	1.7	6.7	6.0	(7)(5)
2C (med α) C	0.7	0.43	1.8	0.11	340	1.2	1.7	7.1	8.0	(8)
3A (hi α) A	0.29	0.62	0.55	0.11	170	0.3	1.7	4.2	4.3	(5)(6)(2)
3B (hi α) B	0.29	0.62	0.55	0.11	260	0.8	1.7	4.7	6.5	(7)(6)
3C (hi α) C	0.29	0.62	0.55	0.11	340	1.2	1.7	5.1	6.5	(6)(7)
4A (ex hi α) A	0.14	1.0	0.17	0.11	170	0.3	1.7	3.3	5.0	(5)
IA (N/α = 3)	0.14	0.83	0.28	0.11	170	0.4	1.7	3.7	5.3	(6)(4.5)
6A (lo q) A	1.7	0.13	3.3	0.04	200	0.2	2.0	9.1	9.0	(10)(8)
6C (lo q) C	1.7	0.13	3.3	0.04	410	0.6	2.0	9.5	9.5	(10)(9)
7A (med q) A	0.71	0.3	3.0	0.07	180	0.3	1.9	7.6	4.5	(4.5)
7B (med q) B	0.71	0.3	3.0	0.07	270	0.5	1.9	7.8	6.0	(5)(7)
7C (med q) C	0.71	0.3	3.0	0.07	360	0.8	1.9	8.1	6.0	(6)
8A (hi q) A	0.46	0.72	0.66	0.14	180	0.5	1.6	4.5	3.1	(3)(4.5)(4.1)(1)
8B (hi q) B	0.46	0.72	0.66	0.14	260	1.0	1.6	5.0	2.5	(3)(2)
8C (hi q) C	0.46	0.72	0.66	0.14	340	1.5	1.6	5.5	5.0	(5)

Table 17 (Cont'd)
LARGE AIRCRAFT PROGRAM - TIME DOMAIN CRITERIA APPLICATION

$$PHQR' = 1.7\ \dot{\alpha}' - 1.44\ N'_{z_p} + 0.55\ T'_{\alpha} + TD' + \dot{q}' + 2.0$$

(where TD' and $\dot{q}'$ used sensitivity factors adjusted to wheel controllers)

CONFIGURATION	1.72 $\dot{\alpha}'$	1.44 N'_{z_p}	0.55 T'_{α}	$\dot{q}$/1b	TD_q	TD'	$\dot{q}'$	PHQR'	AHQR	HQR
9A (med α) A	0.65	0.72	1.27	0.11	170	0.34	1.7	5.2	4.5	(4.5)
9B (med α) B	0.65	0.72	1.27	0.11	260	0.83	1.7	5.7	5.0	(5)(5)
9C (med α) C	0.65	0.72	1.27	0.11	340	1.2	1.7	6.1	6.0	(6)
10A (hi α) A	0.27	1.1	0.39	0.11	180	0.4	1.7	3.7	4.0	(4)(6)(4)(3)
10B (hi α) B	0.27	1.1	0.39	0.11	260	0.83	1.7	4.1	5.0	(5)
10C (hi α) C	0.27	1.1	0.36	0.11	340	1.2	1.7	4.5	5.0	(5)(5)
11A (hi q) A	0.61	1.0	0.6	0.14	180	0.5	1.6	4.3	3.0	(3)
SAT										
12A (med α) A	0.73	0	3.3	0.11	180	0.4	1.7	8.1	10	(10)
12B (med α) B	0.73	0	3.3	0.11	270	0.83	1.7	8.6	10	(10)(10)
13A (hi α) A	0.34	0	3.3	0.11	180	0.4	1.7	7.7	8.3	(9)(5)(5)(6)(4)
13B (hi α) B	0.34	0	3.3	0.11	260	0.83	1.7	8.2	10	(9)
14A (med q) A	0.71	0	3.3	0.06	180	0.2	2.0	8.2	9	(9)
15A (hi q) A	0.48	0	3.3	0.13	180	0.46	1.6	7.8	5.8	(9)(8)(6)(4)
15C (hi q) C	0.48	0	3.3	0.13	430	1.9	1.6	9.3	9	(10)
16A (exhi q) A	0.36	0.1	0.9	0.29	180	1.0	0.8	5.2	4	(4)

6.1.4.5 NT-33A/Ames Study Results

The NT-33A/Ames Study is a yet unpublished program sponsored by NASA Ames to investigate differences between ground and in-flight simulation pilot ratings. The test matrix consisted of three configurations from the LAHOS program (Reference 1), 2-1, 2-7, and 6-1. In addition, two levels of time delay (108 ms and 144 ms) were added to the inherent time delay of 100 ms for Configuration 2-1 for a total of five (5) configurations.

This data provided an excellent check on previous testing of the criteria (Figure 42) on LAHOS configurations. However, in this case there were numerous evaluations by different pilots for each configuration. (A comparison of predicted HQR (PHQR') values from Figure 42 will show slight variation due to small differences in the time histories from the Ames program and the LAHOS work.)

As in the previous LAHOS application the criteria used TD' and $\dot{q}'$ parameters adjusted for center stick controller sensitivities. The results are plotted on Figure 44 with the supporting data in Table 18. The results are summarized by:

CONFIGURATIONS PREDICTED TO WITHIN 1 HQR	CONFIGURATIONS PREDICTED TO WITHIN 2 HQR	CONFIGURATIONS PREDICTED BY FLYING QUALITIES LEVEL OR WITHIN 1 HQR
4 of 5 or 80%	5 of 5 or 100%	4 of 5 or 80%

These results are a significant improvement over the previous testing against LAHOS configuration. The author feels that this is due in large part to the relatively high number of repeats which was made possible by a smaller test matrix.

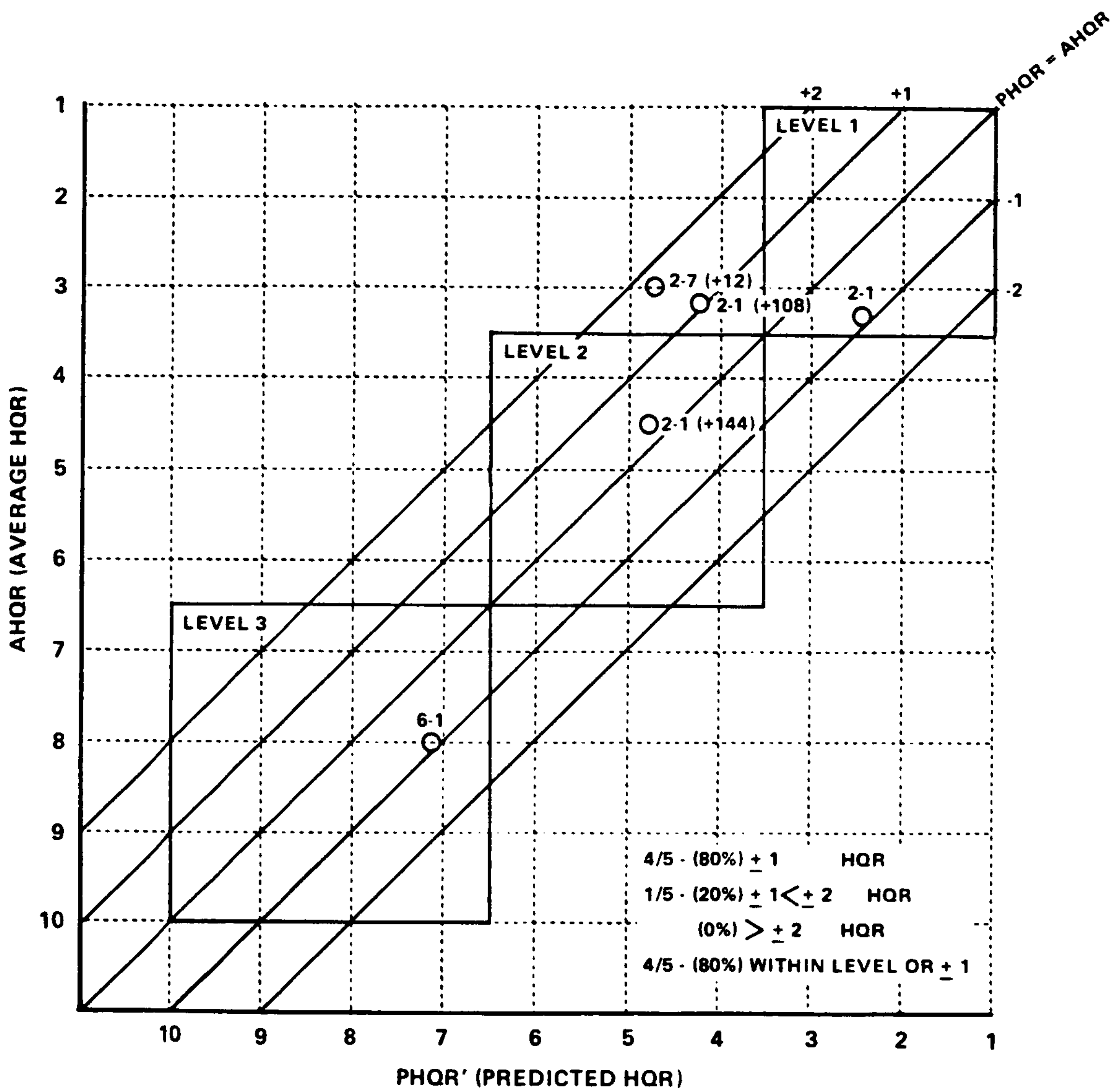


Figure 44 REFINED TIME DOMAIN CRITERIA APPLIED TO THE NT-33/AMES STUDY (NOT YET PUBLISHED)

Table 18
T-33/AMES STUDY - TIME DOMAIN CRITERIA APPLICATION

$(PHQR' = 1.7\,\dot{\alpha}' - 1.44\,N'_{z_p} + 0.55\,T'_{\alpha} + TD' + \dot{q}' + 2.0)$

(where TD' and $\dot{q}'$ used sensitivity factors adjusted for center stick controllers)

CONFIG	$1.72\,\dot{\alpha}'$	$1.44\,N'_{z_p}$	$0.55\,T'_{\alpha}$	$\dot{q}/lb$	TD_q	TD'	$\dot{q}'$	PHQR'	AHQR	HQR	$1/\tau_{\theta_2}$
2-1	0	0	0	2.3	100	0	0.4	2.4	3.25	(2)(2)(3)(3)(4)(4)(5)	0.7
2-1(+108)	0	0	0	2.3	208	1.8	0.4	4.2	3.14	(2)(2)(3)(3)(3)(4)(5)	0.7
2-7(+12)	0	0	0.15	1.9	230	1.8	0.75	4.7	3.0	(2)(2)(2)(3)(4)(5)	0.7
2-1(+144)	0	0	0	2.3	244	2.4	0.4	4.8	4.5	(2)(3)(3)(6)(6)(7)	0.7
6-1	0.1	0	1.5	1.8	450	1.8	1.8	7.2	8.1	(6)(7)(8)(9)(9)(10)	0.7

6.1.4.6 SST Program and X-29 Data Results

The majority of SST data (Reference 12) resulted from divergent configurations that are inappropriate for the criteria, however, three configurations were stable and were applied to the criteria. (These three configurations were also tested on the original time domain criteria.) The refined criteria applied in this case used wheel controller sensitivity parameters.

Some unpublished X-29 data was available for comparison and was applied to the criteria using center stick controller sensitivity parameters.

The results are plotted on Figure 45 with the supporting data in Table 19. The results are summarized by:

CONFIGURATIONS PREDICTED TO WITHIN 1 HQR	CONFIGURATIONS PREDICTED TO WITHIN 2 HQR	CONFIGURATIONS PREDICTED BY FLYING QUALITIES LEVEL OR WITHIN 1 HQR
5 of 5 or 100% (Previous Criteria Results 2 of 3 or 67%)	5 of 5 or 100% (Previous Criteria Results 3 of 3 or 100%)	5 of 5 or 100% (Previous Criteria Results 3 of 3 or 100%)

6.1.4.7 Criteria Testing Summary

The overall results of the time domain criteria testing are presented in Table 20. A total of 129 configurations were tested, of these, the pilot ratings of 77 (60%) were predicted to within one HQR, 113 (88%) were predicted to within two HQR and 16 (12%) had predictions that missed by more than two HQR. Predictions by flying qualities level, or within one HQR, were accurate on 104 of the 129 (or 81%).

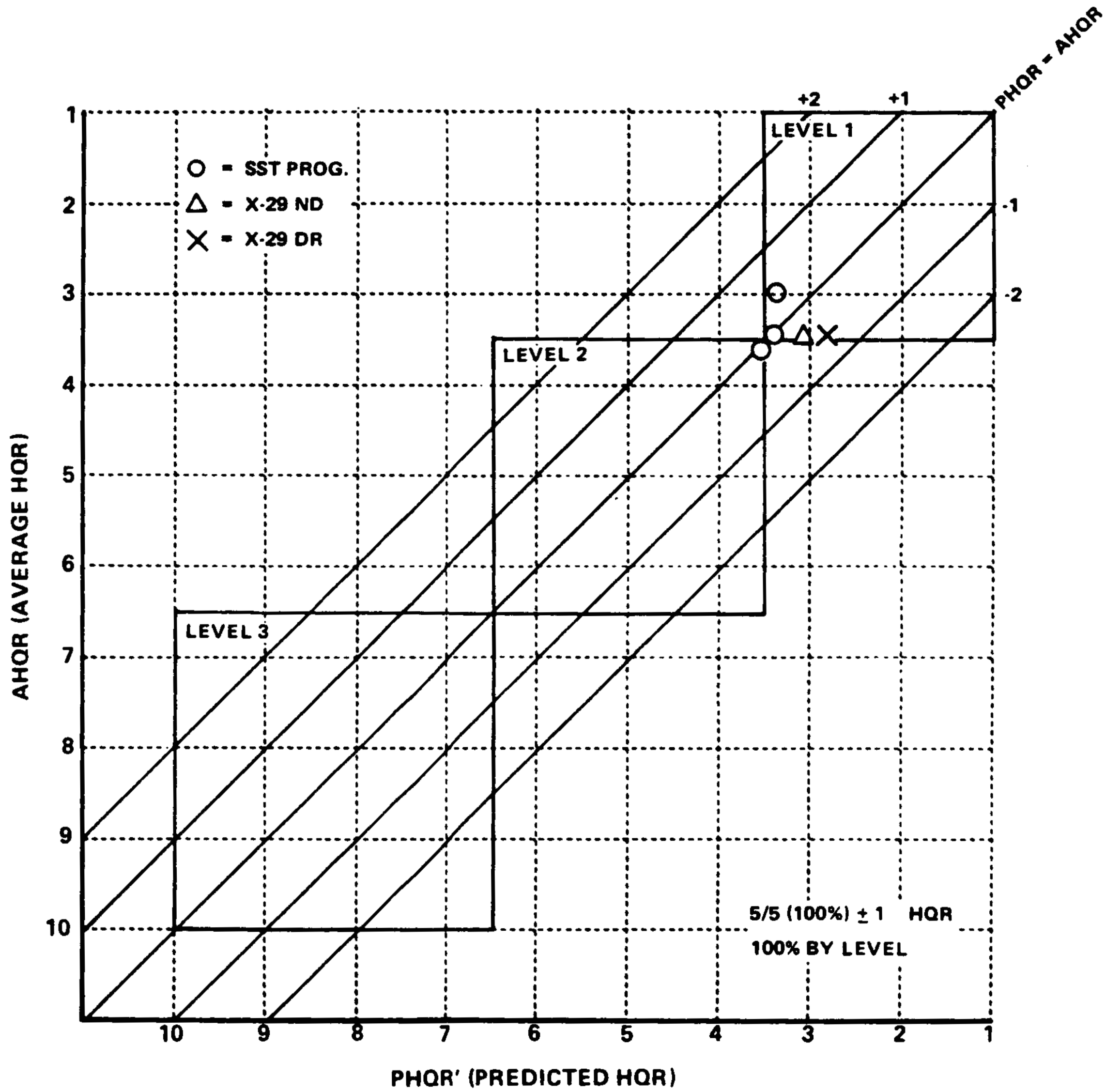


Figure 45 REFINED TIME DOMAIN CRITERIA APPLIED TO THE SST PROGRAM AND X-29 UN-PUBLISHED DATA

Table 19

SST PROGRAM AND X-29 DATA - TIME DOMAIN CRITERIA APPLICATION

$$(PHQR' = 1.7\,\dot{\alpha}' - 1.44\,N'_{z_p} + 0.55\,T'_{\alpha} + TD' + \dot{q}' + 2.0)$$

(where TD' and q' adjusted for wheel controllers for SST data and adjusted for center stick controller for X-29 data

CONFIGURATION	$1.72\,\dot{\alpha}'$	$1.44\,N'_{z_p}$	$0.55\,T'_{\alpha}$	$\dot{q}/lb$	TD_q	TD'	$\dot{q}'$	PHQR'	AHQR	HQR
SST										
1	0	0.52	0.17	0.15	150	0.33	1.5	3.5	3.0	
13	0.1	0.72	0.22	0.15	150	0.33	1.5	3.4	3.5	
20	0.14	0.72	0.26	0.15	150	0.33	1.5	3.5	3.6	
X-29										
N.D.	0.22	0.33	0.2	2.1	120	0.3	0.6	3.0	3.4	
D.R.	0.22	0.33	0.2	2.1	120	0.3	0.6	3.0	3.4	

X-29: $TD'_q = 0.02 \left| \dfrac{TD_{\dot{q}} - 100}{2.8} \right| \dot{q}/lb \qquad \dot{q}/lb' = \left| \dfrac{\dot{q}/lb - 2.8}{1.2} \right|$

Table 20
REFINED TIME DOMAIN CRITERIA TESTING RESULTS

PROGRAM	NUMBER OF CONFIGURATIONS	CONFIGURATIONS PREDICTED WITHIN 1 HQR (No./%)	CONFIGURATIONS PREDICTED WITHIN 2 HQR (No./%)	CONFIGURATIONS MISSED BY MORE THAN TWO HQR	CONFIGURATIONS PREDICTED BY FLYING QUALITIES LEVEL OR WITHIN ONE HQR (No./%)
LANDING PROGRAM	26	20/77%	25/96%	1/4%	26/100%
PITCH RATE PROGRAM	27	17/63%	24/89%	3/11%	21/78%
LAHOS PROGRAM	32	13/41%	23/72%	9/28%	24/75%
LARGE AIRCRAFT PROGRAM	34	18/53%	31/91%	3/9%	24/71%
NT-33/AMES PROGRAM	5	4/80%	5/100%	-	4/80%
SST PROGRAM	3	3/100%	3/100%	-	3/100%
X-29 PROGRAM	2	2/100%	2/100%	-	2/100%
TOTAL	129	77/60%	113/88%	16/12%	104/81%

6.2 FREQUENCY DOMAIN RESULTS AND ANALYSES

In order to provide additional insight into the pilot rating trends obtained in this program, a number of frequency domain flying qualities criteria were applied to the configurations of this study. Linear models of these configurations were used to generate pitch rate, pitch attitude, and attitude rate closed-loop frequency response data. These data were then used to assess the utility of a particular criterion with respect to predicting the flight results. It should be noted, however, that the following analyses do not provide an in-depth assessment of each configuration's flying qualities. Instead, they serve as an attempt to describe the overall pilot rating trends obtained with this data base.

Four frequency domain predictive techniques were considered; lower order eqivalent systems, pitch attitude bandwidth, altitude rate bandwidth, and Neal-Smith pitch attitude pilot lead compensation.

6.2.1 Low-Order Equivalent Systems

The configurations of this study were evaluated using low-order equivalent systems analysis (Reference 19). Closed-loop frequency responses of each configuration were fit with one of the following low-order models:

$$\frac{q}{F_{es}} = \frac{K_q(s + 1/\tau_{\theta_2})e^{-\tau s}}{s^2 + 2\zeta_{sp}\omega_{sp}s + \omega_{sp}^2} \qquad \frac{\alpha}{F_{es}} = \frac{K_\alpha(s + 1/\tau_\alpha)e^{-\tau s}}{s^2 + 2\zeta_{sp}\omega_{sp}s + \omega_{sp}^2}$$

(Frequency range = 0.1 - 10.0 rad/sec)

Selection of a low-order model for a given configuration was based upon its high-order, closed-loop frequency response shape. Configurations without phugoid resonant peaks in their pitch rate frequency response were matched to the pitch rate low-order transfer function with a fixed value of $1/\tau_{\theta_2}$. Those configurations exhibiting phugoid resonant peaks in their pitch rate frequency response were matched to the low-order alpha transfer function. This stemmed from the fact that the equivalent system models did not include the phugoid response in their transfer functions and any phugoid dynamics of significant magnitude would result in unacceptably high levels of mismatch.

The low-order equivalent system results for the configurations of this study are presented in Table 21. Although satisfactory low-order fits were made to most of the configurations, the results did not adequately discriminate between the good and bad configurations. All equivalent short period frequencies were Level 1 (Reference 7). The equivalent short period damping ratios were Level 1, with the exception of the overdamped configurations (2, 4, 6, 8, and 10) which were Level 2. The equivalent time delays were primarily in the Level 2 range, except for the time delay configurations with additional delays which were Level 3 and below (Level 4).

The results in Table 21 were to be expected, since the configurations of this study were originally designed to be low-order models. In addition, the low-order models did not include the phugoid response. The only additional insight is provided by the equivalent time delay results, which show degradation of all configurations to the Level 2 range in the best cases. This does not correlate well with the documented pilot ratings comments. Overall, only 38% of the configurations were accurately predicted by level using the equivalent system approach.

6.2.2 Pitch Attitude Bandwidth Criterion

The pitch attitude bandwidth criterion (Reference 20) was used to provide additional insight into the pitch characteristics of the configuration in this study. The criterion was applied to the closed-loop frequency response of each configuration with the bandwidth defined as the crossover frequency at which the phase margin is 45° or the gain margin is 6 dB, whichever frequency is lower. In addition, an estimate of the time delay of each configuration is calculated to be:

$$\tau = -(\phi + 180^\circ)/(57.3\omega)$$

where ω is defined to be two times the frequency at 180° and ϕ is the phase at ω. This criterion was formulated using the configurations from Reference 1.

The pitch attitude bandwidth results for the configurations in this study are presented in Table 22. The level ratings in Table 22 were based upon the level boundaries established in Reference 20, which were determined using the pilot ratings for the fighter aircraft configurations from Reference 1. As a result, only 44% of the configurations were accurately predicted by

Table 21

LOW-ORDER EQUIVALENT SYSTEM RESULTS

CONFIG NO	TRANSFER FUNCTION FIT	NUMERATOR TERM*	ZETA	LEVEL	OMEGA (R/S)	LEVEL	TIME DELAY** (SEC)	LEVEL	OVERALL PREDICTED LEVEL	PILOT RATING LEVEL
1	α	$1/\tau_\alpha = 4.1$	.71	1	2.05	1	0.145	2	2	1
2	q	$1/\tau_{\theta 2} = .5$	1.97	2	2.08	1	0.145	2	2	1
3	q	$1/\tau_{\theta 2} = .5$	.69	1	2.00	1	0.147	2	2	2
4	α	$1/\tau_\alpha = 3.3$	1.97	2	1.81	1	0.142	2	2	1
5	α	$1/\tau_\alpha = 4.5$	.71	1	2.06	1	0.145	2	2	2
6	q	$1/\tau_{\theta 2} = .9$	1.32	2	1.98	1	0.146	2	2	1
7	q	$1/\tau_{\theta 2} = .9$	.70	1	2.00	1	0.147	2	2	2
8	α	$1/\tau_\alpha = 11.2$	2.00	2	3.63	1	0.150	2	2	1
9	q	$1/\tau_{\theta 2} = .5$	.69	1	2.00	1	0.147	2	2	1
10	q	$1/\tau_{\theta 2} = .5$	2.00	2	1.97	1	0.145	2	2	1
11-12	Reasonable fits not possible								3	3
13	α	$1/\tau_\alpha = 4.1$	.71	1	2.10	1	0.145	2	2	2
14	q	$1/\tau_{\theta 2} = 2.0$	1.00	1	1.99	1	0.146	2	2	2
16	α	$1/\tau_\alpha = 4.6$	.69	1	3.22	1	0.142	2	2	1
17	q	$1/\tau_{\theta 2} = .72$	1.10	1	2.00	1	0.170	2	2	1
18-20	Reasonable fits not possible								3	1
B***	q	$1/\tau_{\theta 2} = .75$	.76	1	1.90	1	0.148	2	2	1
21	q	$1/\tau_{\theta 2} = .75$	.76	1	1.90	1	0.248	3	3	2
22	q	$1/\tau_{\theta 2} = .75$	.76	1	1.90	1	0.348	4****	4	3
23	q	$1/\tau_{\theta 2} = .75$	.76	1	1.90	1	0.148	2	2	2
24	q	$1/\tau_{\theta 2} = .75$	.76	1	1.90	1	0.248	3	3	1
25	q	$1/\tau_{\theta 2} = .75$	.76	1	1.90	1	0.348	3	3	2
26	q	$1/\tau_{\theta 2} = .75$	.76	1	1.90	1	0.148	2	2	1
27	q	$1/\tau_{\theta 2} = .75$	.76	1	1.90	1	0.248	3	3	2
28	q	$1/\tau_{\theta 2} = .75$	.76	1	1.90	1	0.348	4	4	3
17+L/L	q	$1/\tau_{\theta 2} = .75$	.78	1	2.90	1	0.160	2	2	1
22A	q	$1/\tau_{\theta 2} = .75$	.78	1	2.90	1	0.360	4	4	3
25A	q	$1/\tau_{\theta 2} = .75$	.78	1	2.90	1	0.360	4	4	2
28A	q	$1/\tau_{\theta 2} = .75$	.78	1	2.90	1	0.360	4	4	3

* $1/\tau_{\theta 2}$ terms fixed, $1/\tau_\alpha$ terms free.

** Referenced to stick force. Feel system and actuator included.

*** Configurations B, 21-28 fit in the frequency range of 0.3-10.0 rad/sec.

**** Level 4 is worse than Level 3

Table 22

PITCH ATTITUDE BANDWIDTH CRITERION RESULTS

CONFIG NO	BANDWIDTH (Phase Margin) (rad/sec)	BANDWIDTH (Gain Margin) (rad/sec)	TIME DELAY (sec)	PREDICTED LEVEL*	PILOT RATING LEVEL
1	2.36	2.86	0.114	2	1
2	3.01	3.65	0.109	1	2
3	2.38	2.87	0.114	2	2
4	2.96	3.65	0.109	2	1
5	2.07	2.59	0.114	2	1
6	2.24	3.13	0.109	2	1
7	2.13	2.65	0.114	2	2
8	2.12	3.08	0.109	2	1
9	2.38	2.87	0.114	2	1
10	3.01	3.65	0.109	1	1
11	1.16	1.44	0.226	3	3
12	1.23	1.11	0.308	3	3
13	1.53	2.08	0.119	2	2
14	1.36	2.38	0.108	2	2
16	3.06	3.24	0.117	2	1
17	2.44	2.77	0.126	2	1
18	2.48	2.81	0.126	2	1
19	2.52	2.84	0.126	2	1
20	2.60	2.90	0.126	2	1
B	2.20	2.71	0.114	2	1
21	2.20	2.71	0.214	3	2
22	2.20	2.71	0.314	3	3
23	2.20	2.71	0.114	2	2
24	2.20	2.71	0.214	3	1
25	2.20	2.71	0.314	3	2
26	2.20	2.71	0.114	2	1
27	2.20	2.71	0.214	3	2
28	2.20	2.71	0.314	3	3
17+L/L	3.13	3.21	0.127	2	1
22A	3.13	3.21	0.317	3	3
25A	3.13	3.21	0.317	3	2
28A	3.13	3.21	0.317	3	3

* Level determined by boundaries established in Reference 18 for category flight phases.

level. Considering the significant differences between fighter and transport aircraft, the pitch attitude bandwidth results in Table 22 were combined with similar results from the configurations of Reference 5 to produce suggested boundaries for transport aircraft configurations. These results are presented in Figure 46 and include 58 generic transport configurations between the two programs. The suggested level boundaries in Figure 46 result in 70% of the configurations being correctly predicted by level, a marked improvement from the boundaries used in Reference 20. The location of the level boundaries in Figure 46 relative to those used for fighter aircraft in Reference 20 bring out two important points. First, pilots flying transport aircraft can accept a lower bandwidth in pitch and still achieve desired performance. Second, transport aircraft can have higher values of allowable time delay relative to fighter aircraft, however, this is dependent on the types of controllers used and sensitivities selected.

6.2.3 Altitude Rate Bandwidth Criterion

Control of altitude rate in the flared landing task was analyzed for the configurations of this study through a single-loop closure technique used in Reference 5. A single-loop closure was performed on altitude rate around the following pilot model:

$$Y_{p_h} = K_{p_h} e^{-\tau s} \quad (\tau = 0.25 \text{ sec})$$

The closed-loop performance was determined through the use of a Nichols chart over the frequency range of 0.1 to 10.0 rad/sec. The altitude rate bandwidth was the frequency at which 3 dB of closed-loop resonance corresponded with -90° of closed-loop phase. This procedure was performed on a Nichols chart through the manipulation of the pilot's bandwidth and gain. This technique, although obtaining limited success in Reference 5, showed promise when applied to the configurations of this study.

The altitude rate bandwidth and corresponding pilot gain results are presented in Table 23. A correlation was made between the amount of pilot gain required to achieve the stated closed-loop performance and the altitude bandwidth. These results are presented in Figure 47. As expected, configurations that allow the pilot to increase his gain in the control of altitude rate (for a given bandwidth) were generally considered to be Level 1. Figure

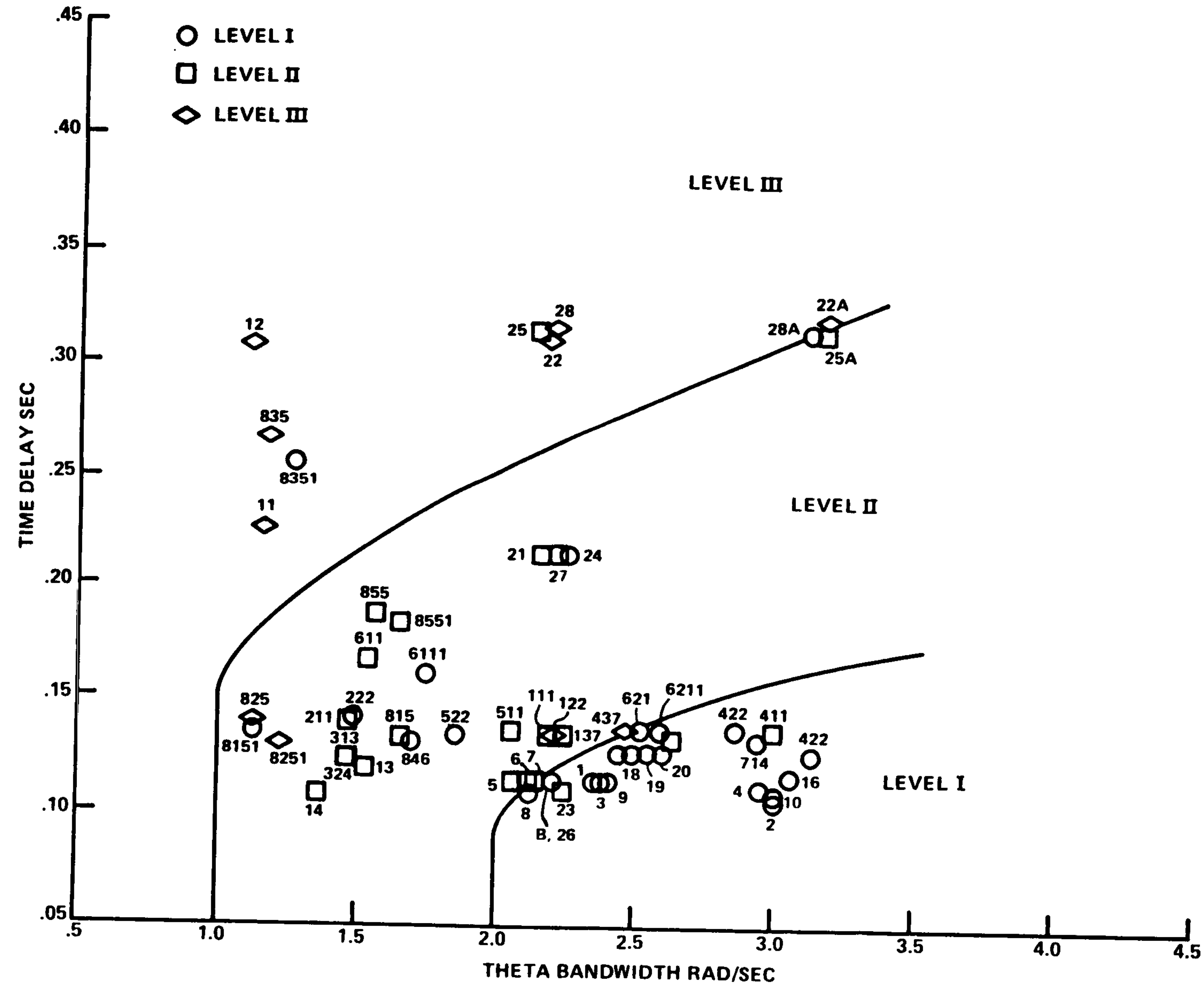


Figure 46 THETA BANDWIDTH RESULTS (REF. 5 CONFIGS. INCLUDED)

Table 23

ALTITUDE RATE BANDWIDTH RESULTS

CONFIGURATION NO.	PILOT GAIN	BANDWIDTH (rad/sec)
1	2.32	1.65
2	4.05	1.43
3	2.40	1.68
4	3.57	1.35
5	1.49	1.35
6	1.75	1.13
7	1.58	1.43
8	1.49	1.00
9	2.48	1.65
10	4.05	1.38
11	0.61	0.83
12	0.78	1.05
13	0.68	1.18
14	0.78	1.03
16	3.55	1.93
17	1.56	0.78
18	1.71	0.83
19	1.90	0.95
20	2.17	0.95
B	1.97	0.90
21	1.74	0.83
22	1.54	0.75
23	3.30	0.90
24	2.90	0.83
25	2.58	0.75
26	1.32	0.90
27	1.17	0.83
28	1.02	0.75
17+L/L	2.10	1.20
22A	1.60	0.98
25A	1.30	0.90
28A	1.71	1.00

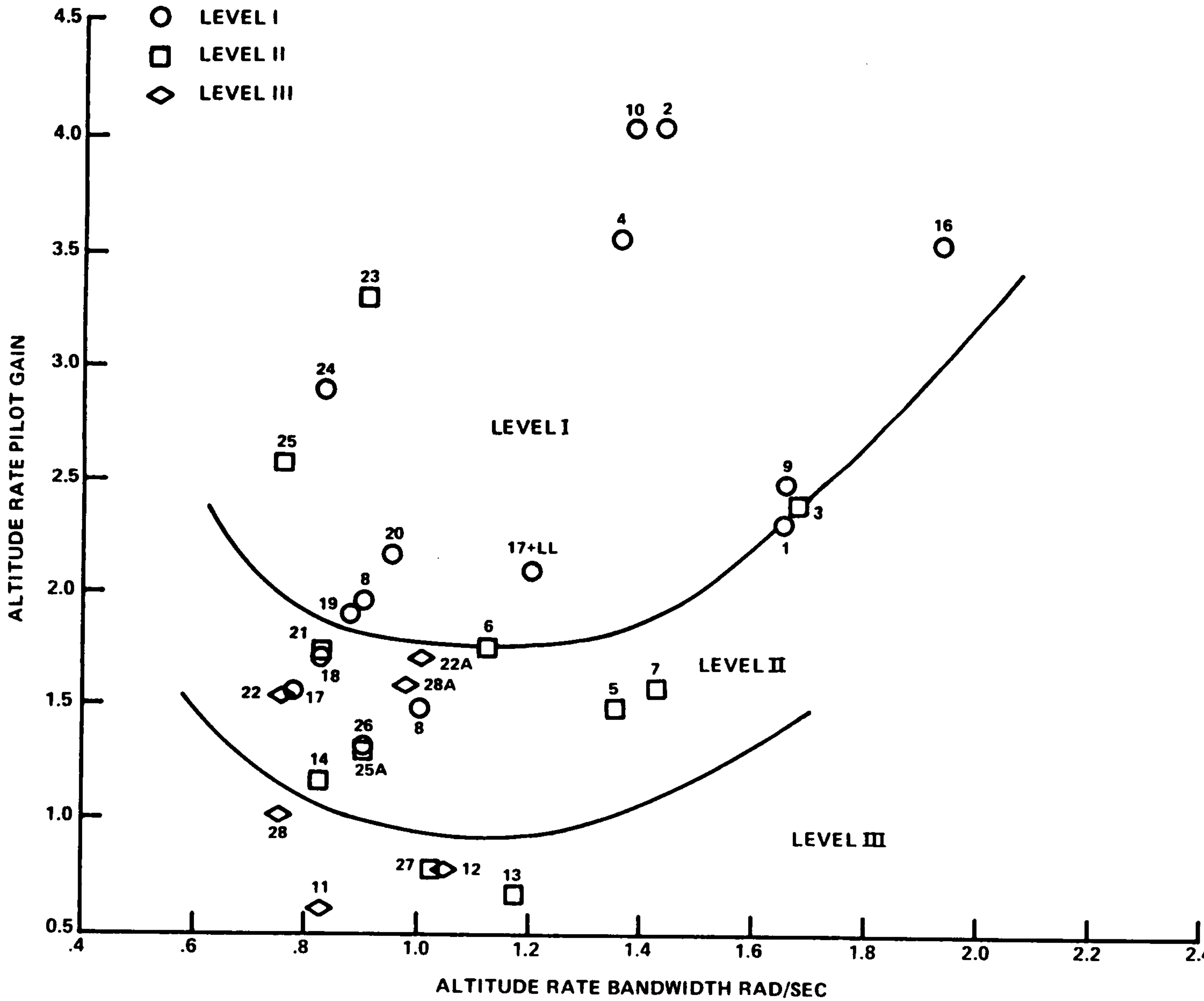


Figure 47 ALTITUDE RATE BANDWIDTH RESULTS (CURRENT PROGRAM)

47 also shows proposed level boundaries that result in 63% of the configurations being accurately predicted by level, however, only 53% of the time delay/sensitivity configurations were predicted by level using this technique.

6.2.4 Neal-Smith Analysis

The Neal-Smith criterion (Reference 21) was applied to the data base and correlated with the results from Reference 5 to provide insight into the pitch characteristics of generic transport aircraft. The criterion is based upon a single-loop closure performed on pitch attitude using the following pilot model:

$$Y_{p_\theta} = K_{p_\theta} e^{-\tau s} (\tau_{lead} s + 1)/(\tau_{lag} s + 1)$$

The pilot model operates on a pitch-attitude error signal that is the difference between the commanded attitude and the aircraft attitude. The pilot, through the parameters he is observing, tries to achieve a certain "standard of performance" which is defined by a specified closed-loop bandwidth. The bandwidth is defined by the 90° closed-loop phase requirement. At frequencies below the bandwidth, the pilot attempts to minimize steady-state pitch attitude tracking errors as defined by a minimum low-frequency droop (typically no more than -3 dB). The pilot also attempts to minimize the closed-loop resonant peak, which minimizes oscillatory tendencies in pitch attitude. The pilot model is adjusted so that the -3 dB droop and the -90° of the closed-loop phase conditions are met for a given bandwidth while the closed-loop resonance is minimized. These parameters then provide a measure of the compensation with which the pilot closes the loop in pitch attitude. After the closed-loop conditions are met, closed-loop resonance and pilot compensation are plotted on a Neal-Smith parameter plane and correlated with the pilot ratings for the flared landing task.

The results of the Neal-Smith criterion are presented for each configuration in Appendix F. A bandwidth frequency of 2.0 rad/sec appeared to best represent the configurations of this study (in terms of predicting the flying qualities levels). This correlates with the pitch attitude bandwidth results in Reference 5. Analysis of the data showed that variations in pilot rating appeared to be dependent upon the amount of pilot lead or lag required to achieve the closed-loop requirements. The closed-loop resonance at 2.0

rad/sec was Level 1 for all configurations and was not a factor in the pilot ratings.

A distinct correlation was made between pitch attitude pilot compensation and pilot ratings, as shown in Figure 48. The results of Figure 48 show that pilot lead compensation less than or equal to approximately 25° is required for Level 1 flying qualities of transport aircraft in the flared landing task. In addition, the results show a linear degradation in pilot ratings for increasing pilot compensation. Configurations from Reference 5 and selected configurations for Reference 4 support this trend. However, of the time delay/sensitivity configurations only 54% were predicted by level and 54% predicted to within ±1 HQR. This trend and correlation was not evident in the Neal-Smith analyses performed in References 4 and 5. It is also interesting to note that the 25° "elbow" for Level 1 flying qualities in Figure 48 is consistent with the findings in Reference 22 regarding pitch attitude pilot compensation. Although pitch attitude is not the only control variable in the flared landing task, it is an essential part of any closed-loop flared landing technique employed by pilots. Through the combination of configurations selected from these three data bases, 73% of the configurations were predicted by level and 65% were predicted within 1 pilot rating.

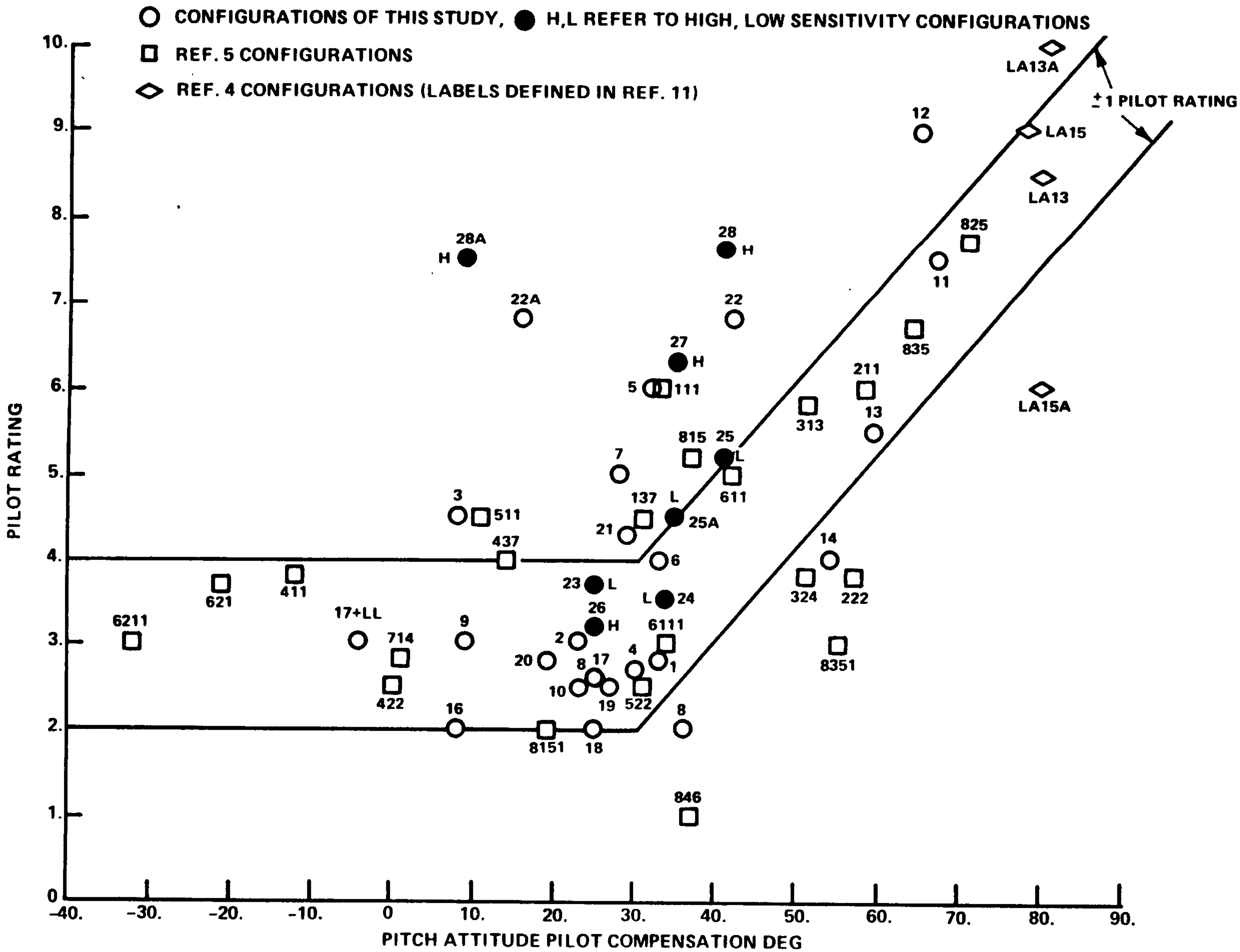


Figure 48 NEAL-SMITH RESULTS (PITCH ATTITUDE BANDWIDTH = 2.0 RAD/SEC)

6.3 COMPARISON OF TIME DOMAIN CRITERIA AND FREQUENCY DOMAIN CRITERIA PERFORMANCE

The time domain criteria was tested against seven sets of data and 129 configuration which included fighter type aircraft with stick controllers. The frequency domain criteria were tested against one to three sets of data and 58 configurations, the majority of which were contained in the present landing program and the previous pitch rate program and which consisted of medium transport aircraft with wheel controllers. All of the 58 frequency domain configurations were included in the 129 configurations of the time domain analysis. In order to obtain a direct comparison between criteria the specific 58 frequency domain configurations were pulled from the time domain results. The comparison of the frequency domain and time domain criteria results of these configurations are shown in Table 24.

The low-order equivalent systems technique predicted 38% of the landing program configurations by level as compared to 100% prediction by level of the time domain criteria. The low-order equivalent systems method does not appear applicable to the flared landing task.

The pitch attitude bandwidth technique was only 44% successful in level prediction using previous boundaries but when new boundaries were drawn based on 58 configurations of the landing and pitch rate programs 70% of the configurations fell inside the revised boundaries. This result is promising, however, washout prefilter effects could not be predicted and more data is required in the area of high bandwidth and higher time delays to more accurately describe the boundaries. By comparison the time domain criteria predicted 89% of these configurations by level.

The altitude rate bandwidth was applied to the configurations of the landing program. New boundaries were established which allowed 63% of the configurations to fall within levels, however, time delay and sensitivity configurations were not accurately predicted and 3 of 6 Level 3 configurations were missed. By comparison the revised time domain criteria predicted 100% of the configurations by level.

Table 24

COMPARISON OF FREQUENCY DOMAIN AND TIME DOMAIN PREDICTIVE TECHNIQUES

PREDICTIVE METHOD	LANDING PROGRAM (32 Configs Total)	PITCH RATE PROGRAM (27 Configs Total)	LARGE AIRCRAFT PROGRAM (34 Configs Total)
Low Order Equivalent Systems	38% Predicted by Level (32 configs tested)	- - -	- - -
Pitch Attitude Bandwidth Using Fighter Boundaries	44% Predicted by Level (32 configs tested)	- - -	- - -
Pitch Attitude Bandwidth Using New Boundaries	70% Predicted by Level (58 configs tested) Low frequency without Filters not accurately predicted		- - -
Altitude Rate Bandwidth Using New Boundaries	63% Predicted by Level (32 configs. tested) Time Delay and Sensitivity not accurately predicted		- - -
Neal-Smith Pilot Lead Compensation Using Suggested Boundaries	73% Predicted by Level 65% within ±1 HQR 19% missed by >2 HQR 4 Level 3's missed (26 configurations tested) Time Delay and Sensitivity not accurately predicted	77% Predicted by Level 60% within ±1 HQR 14% missed by >2 HQR no Level 3's missed (22 configurations tested)	75% Predicted by Level 50% within ±1 HQR 25% missed by >2 HQR no Level 3's missed (4 configurations tested)
Refined Time Domain Criteria	100% Predicted by Level 77% within ±1 HQR 4% missed by >2 HQR no Level 3's missed (26 configurations tested)	78% Predicted by Level 63% within ±1 HQR 11% missed by >2 HQR 2 Level 3's missed (22 configurations tested)	75% Predicted by Level 50% within ±1 HQR 0% missed by >2 HQR no Level 3's missed (4 configurations tested)

The Neal-Smith criteria which was applied to configurations from the landing program (26 configurations), pitch rate program (22 configurations) and the large aircraft program (4 configurations) showed significant correlation with pitch attitude pilot compensation. Boundaries could be drawn (Figure 48) that allowed correct predictions of 75% of the configurations by level and 62% within ±1 HQR. This is the most promising of the frequency domain techniques tested, however, time delay/sensitivity configurations of the landing program were not accurately predicted. By comparison the revised time domain criteria predicted 88% of the above configurations by level, 69% within ±1 HQR and of the time delay/sensitivity configurations 100% were predicted by level and 85% were predicted to within ±1 HQR.

In the above comparisons the frequency domain criteria were applied to primarily two basic data sets containing 58 configurations. Assumptions were then made; i.e., Neal-Smith pilot bandwidth, etc., and boundaries were drawn that best matched the criteria with the data, i.e., empirical boundary definition. These results were then compared to the revised time domain criteria which was applied to the same data. These comparisons are shown in Table 24. The time domain criteria was empirically developed from the pitch rate program, and the revised time domain criteria from the landing program. The criteria was then tested on 5 other data sets (Table 20) for a total of 129 configurations that included fighter and transport aircraft. As described in the previous paragraph on a one to one comparison the revised time domain criteria was more accurate than the Neal-Smith pilot lead compensation method (the most promising of the frequency domain methods) by all comparison metrics and considerably more accurate with time delay and sensitivity configurations. In fact, there has been no other criteria observed that will account for sensitivity.

When the refined time domain criteria was applied to the additional data sets the overall results of 129 configurations listed in Table 20 were:

predicted by level - 81%
predicted within ±1 HQR - 60%
configurations missed by more than 2 HQR - 12%

These total results were more accurate than the best of the frequency domain methods that were empirically fitted to a 58 configuration data base. The time domain criteria has fewer limitations, and is especially accurate for predicting time delay and sensitivity effects.

6.4 WASHOUT INVESTIGATION

Configurations 17 through 20 were evaluated to determine the effect of a washout prefilter on a rate command-attitude hold system. Configuration 17, which was identical to Configuration 1-2-2 of the Reference 5 TIFS/Pitch study, was chosen as a typical rate command system. It had recieved pilot ratings of $5\frac{1}{2}$, 7, and 8 in that reference study. Washout prefilters with frequencies of .05, .1, and .2 rad/sec were added with Configurations 18, 19, and 20, respectively.

It was postulated that Configuration 17 would again receive borderline Level 2-3 ratings as it did in the previous program, and then the potential improvement in flying qualities with various degrees of washout in the command path could be evaluated. As it turned out in this program, this configuration received pilot ratings all between 2 and 4 with an average of 2.6 from the five separate pilot evaluations. The maximum washout configuration (20), with an ω_{WO} = .2 rad/sec, was evaluated by four pilots and also received pilot ratings between 2 and 4 with an average of 2.8. The only significant pilot comments (see Appendix D) were that the no washout configuration (17) held flight path better in the approach phase, but that the maximum washout configuration (20) felt more natural with the required aft forces, in the flare.

The predictive criteria in this study and Reference 5 indicate that the baseline rate command system, Configuration 17, should indeed be rated near the Level 1-2 border as it was in this program. The results from this study also show that the addition of a washout pre-filter does not degrade the flying qualities but just requires the pilot to hold aft forces in the flare. A better investigation into washout filters would require a baseline rate command system that was definitely a Level 2 or 3 configuration.

Calibration steps for Configuration 1-2-2 of Reference 5 were re-examined and found to be the same as those for Configuration 17 of the present study. It is not known why this configuration was rated worse in the previous study, other than the pilot comments indicate a floating overcontrol tendency in the flare which did not seem to bother the pilots in the present study.

6.5 CONCLUSIONS

Conclusions presented here relate to the predictive criteria results. Conclusions from the interpretation of the results using MIL-F-8785(C) as a flight control design criteria are presented in Section 5.4.

6.5.1 Time Delay and Sensitivity Considerations

There is strong and consistent correlation between time delay effect on pilot performance and pitch sensitivity for the flared landing task (observations from other programs and experience in in-flight simulators indicates strongly that a similar correlation exists for the roll axis and other tasks as well). This correlation has been quantified to the extent that the designer now has metrics that will allow him to predict the effects on flying qualities of various time delay and sensitivity values.

The results of this program also call attention to the importance of proper sensitivity selection. Changes in flight control schemes, i.e., addition of prefilters, changes in feedback gains, or any factor that will change the shape of the time response, requires proper adjustments of command gain to maintain sensitivity within desired bounds. For wheel controllers in the flared landing task these bounds are now well defined and the designer has metrics available that allow him to quantify, in flying qualities terms, the effect of design changes. For center stick controllers the bounds are not yet as well defined. For other controllers more data is required.

6.5.2 Time Domain Criteria Considerations

The time domain criteria is considered to be sufficiently refined to be of use to the designers for flying qualities performance predictions of pitch flight control systems in the flared landing task.

The accuracy of the criteria has been tested against a significant number of diverse programs and flight control system designs. In addition, it includes significant closed-loop items that previous criteria could not account for (sensitivity factors, etc.).

The time domain criteria developed from the program requires the following items for HQR predictions:

- computer generated time histories of stick force (F_S), angle of attack (α), pilot station Z-axis acceleration (N_{Z_p}), and pitch rate (q) responses to a step-in, step-out elevator force command.
- a pencil
- a ruler
- a hand calculator
- five minutes of time

6.5.3 Frequency Domain Criteria Considerations

1. Low-order equivalent system results did not adequately discriminate between the good and bad configurations in this study. Only 38% of the 32 configurations of this program were predicted by level. This is largely due to the fact that most of the configurations had significant phugoid response characteristics, and these were not identified by the low-order models.

2. the pitch attitude bandwidth level boundaries were revised to reflect transport aircraft configurations. Using these boundaries 70% of the 58 configurations from this and the pitch rate programs were predicted by level. Configurations with low-frequency washout prefilters were not accurately predicted by this criterion.

3. The altitude rate bandwidth criterion yielded satisfactory results, with 63% of the 32 configurations of this program predicted by level. The characteristics of the time delay configurations in this study were not accurately predicted by this criterion.

4. The Neal-Smith criterion showed the most promising results. 73% of 52 configurations (26 from the landing program, 22 from the pitch rate program, and 4 from the large aircraft program) were predicted by level using this technique. The criterion showed a strong correlation between pilot lead compensation and pilot rating, again with the exception of the time delay configurations.

6.5.4 Overall Predictive Criteria Considerations

The revised time domain criteria provided the most accurate results of all criteria tested. It was tested against more configurations with more diverse characteristics than the other criteria and has fewer limitations. Additionally, it can account for flight control variations and significant in these are time delay and sensitivity. This criteria has matured to the point that it can be used as a predictive tool for the flight control designer.

The Neal-Smith criteria pitch attitude pilot lead compensation technique showed the most promise of the frequency domain criteria, however, it has not been applied to a large enough data base and should be modified to accurately account for sensitivity effects.

6.6 RECOMMENDATIONS

Recommendations presented here relate to the predictive criteria results. Recommendations from the interpretation of the results using MIL-F-8785(C) as a flight control design criteria are presented in Section 5.4.

6.6.1 Time Delay Sensitivity

It has been shown that controller sensitivity has a strong effect on pilot interaction with time delay. Other characteristics in the command path such as feel system frequency, damping, deadband, friction, etc., may have similar effects. A flight program should be conducted to quantify these effects.

6.6.2 Time Domain Criteria

This program has provided quantitative data on wheel controller pitch sensitivity requirements for the flared landing task. Other flight programs should be conducted to investigate sensitivity requirements of wheel, stick, and side stick controllers in both the pitch and roll axes and in other tasks as well as flared landings.

6.6.3 Frequency Domain Criteria

Future efforts in frequency domain analysis should include the pilot's closed-loop gain as an integral part of any predictive criterion. In general, the criteria applied in these analyses did not adequately predict the pilot rating trends obtained from the time delay configurations. It is recommended that a frequency domain criterion to be developed that consistently predicts the results from such configurations.

Section 7

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16. Abstract

An in-flight research study was conducted utilizing the USAF/Total In-Flight Simulator (TIFS) to investigate longitudinal flying qualities for the flared landing approach phase of flight. The purpose of the experiment was to generate a consistent set of data for: (1) determining what kind of commanded response (e.g., angle of attack or pitch rate) the pilot prefers/requires in order to flare and land an airplane with precision, and (2) refining a time history criterion that took into account all the necessary variables and their characteristics that would accurately predict flying qualities. Seven evaluation pilots participated in this experiment representing NASA Langley, NASA Dryden, Calspan, Boeing, Lockheed, and DFVLR (Braunschweig, Germany). The results of the first part of the study provides guidelines to the flight control system designer, using MIL-F-8785-(C) as a guide, that yield the dynamic behavior pilots prefer in flared landings. The results of the second part of the study provides the flying qualities engineer with a newly derived flying qualities predictive tool which appears to be highly accurate. This time-domain predictive flying qualities criterion was applied to the flight data of the present study as well as six previous flying qualities studies, and the results indicate that the criterion predicted the flying qualities level 81 percent of the time and the Cooper-Harper pilot rating, within ±1, 60 percent of the time.

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